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NOMENCLATURE

c	chord, m
C_p	pressure coefficient, $\frac{P_1 - P_\infty}{q_\infty}$
C_L	wing lift coefficient
CLL	section lift coefficient
M	Mach number
P	pressure, N/m ²
q	dynamic pressure, N/m ²
Re	Reynolds number
X	chordwise distance, m
ALPHA	angle of attack, deg
ETA	nondimensional spanwise distance

Subscripts:

l	local
∞	free stream

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A COMPARISON OF THEORETICAL AND EXPERIMENTAL PRESSURE DISTRIBUTIONS FOR TWO ADVANCED FIGHTER WINGS

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SUMMARY

A comparison was made between experimental pressure distributions measured during testing of the Vought A-7 fighter and the theoretical predictions of four transonic potential flow codes. Isolated wing and three wing-body codes were used for comparison. All comparisons are for transonic Mach numbers and include both attached and separate flows.

In general, the wing-body codes gave better agreement with the experiment than did the isolated wing code but, because of the greater complexity of the geometry, were found to be considerably more expensive and less reliable.

INTRODUCTION

The past 10 years has seen the development of numerous transonic-flow analysis codes based on finite difference solutions of the exact potential and small disturbance equations. The most widely used of these codes are those developed by Jameson and Caughey (refs. 1 and 2); FL022, the most reliable of these codes, computes the flow about an isolated wing on a wall. FL022 has been used extensively in the design of advanced wings for transonic aircraft. Data from two such wings designed for the Vought A-7 fighter are compared with theoretical calculations from four different transonic codes to show the strengths and weaknesses of each code. The two wings were designed during a cooperative study involving the Vought Corp., the David Taylor Naval Ship Research and Development Center, and the Ames Research Center. During the cooperative study a wing design code was developed by coupling FL022 with a numerical optimization algorithm (ref. 3). This report will demonstrate that wing design for transonic aircraft by use of isolated wing codes may lead to unsatisfactory results if the aircraft has a low fineness ratio fuselage.

FL022 COMPARISON WITH EXPERIMENT

FL022 is a full potential, non-conservative, transonic analysis code which uses an explicit, square root transformation to map an isolated wing to a "bump" on a plane.

Theoretically predicted pressure distributions from FL022 disagreed significantly from test data at the wing design condition, 0.85 Mach number and $0.30C_L$ (fig. 1). The

wing performed satisfactorily; but when compared with the original A-7, underprediction of shock strength may have resulted in premature drag divergence. Possible causes of this disagreement were identified as code approximations, isolated wing limitations, boundary layer analysis approximations, and test inaccuracies.

A strip boundary layer displacement thickness computed for the design pressure distribution was added to the wing geometry during both design and analysis which introduced an error into the theoretical calculations because the experimental spanwise pressures differ from the design pressures which resulted in an improper displacement thickness. In order to assess this error, FLO22NM predictions were compared with the experiment (fig. 2). FLO22NM combines Jameson's FLO22 code with a Nash-McDonald boundary layer analysis. Displacement thicknesses were computed for each spanwise strip and updated as the solution converged. FLO22NM predictions were somewhat closer to the experiment than FLO22 pressures, but the strong experimental shock was still not predicted. A comparison of FLO22 and FLO22NM pressures is shown in figure 3.

FLO28 COMPARISON WITH EXPERIMENT

FLO28 is a full potential, fully conservative, transonic analysis code developed by Jameson (ref. 3). The code uses the finite volume approach and solves for the flow about a wing in the presence of a finite body. The body is mapped to a vertical slit by use of a generalized Joukowski transformation.

The A-7 body with a 12% thick supercritical wing was analyzed using FLO28 on the NASA Ames CDC 7600 computer. Initial solutions were remarkably bad. Wing pressures were characterized by chordwise oscillations. Comparable irregularities and discontinuities were seen in the wing mapped surface coordinates at the root and tip. Since the correctly input wing-body geometry caused unsmooth transformed coordinates, it was obvious that some compromise in the geometric representation would be required. The body was modified because of its incompatibility with the code approximations. The Joukowski transformation had difficulty mapping the A-7 body which has a nearly rectangular fuselage. Progressive rounding of the body corners had a smoothing effect on the root and tip mapped coordinates and on the resultant pressures. A slight lowering of the A-7 high wing was required by this body modification. These geometry changes smoothed wing pressures, but oscillations were still so bad that it was impossible to compare the theory with experiment.

It seems illogical that smoothing the mapped surface coordinates had not caused a more pronounced improvement in the wing pressures. The code was modified to plot mapped coordinates at every computational station instead of only the root and tip. While the earlier changes had smoothed the root and tip, intermediate span stations were still very irregular - particularly in the 30%-40% semi-span region. While the use of a circular fuselage model seemed the easiest fix, wing height changes were tried first. It was found that even a slight lowering of the wing relative to the body made dramatic improvements in smoothing the mapped coordinates. The best results were achieved with a mid-wing position on the fuselage with slightly rounded corners. Adjusting input geometry to achieve a criteria of smooth mapped coordinates may lack mathematical rigor, but it does produce improved results. Further, it is a relatively easy to use criterion and is not dependent upon agreement of theory with a particular set of test data.

FL022 and early FL028 comparisons with experiment showed better agreement if the theory was run at a slightly higher Mach number than the test data. Other investigators had reached this conclusion. The best agreement with 0.85 Mach number experimental pressures was achieved with theoretical data run at 0.86 Mach number. It was hypothesized that this relatively large Mach shift might indicate a small experimental calibration error. In retrospect, after making comparisons with other codes, this reasoning appears specious.

Once a satisfactory input geometry was achieved, analyses were run at experimental lift conditions. The theory was run at approximately one degree higher angle-of-attack in order to match the test lift. Each analysis was run 50 iterations on the crude and medium grids, and 250 iterations on a fine grid of $160 \times 16 \times 28$. A typical average residual was converged to 2×10^{-6} . Each run required approximately 1700 sec of CDC 7600 time. A few runs were executed an additional 250 iterations on the fine grid, doubling the run time and reducing the average residual to 0.6×10^{-6} , while having little effect on the pressures. The tip shock was strengthened and moved aft very slightly, but pressures over most of the wing were unchanged.

FL028 wing pressure predictions are compared with the experiment at $0.34 C_L$, the wing design condition (fig. 4). Some mild upper surface pressure oscillations are evident from the wing root to mid-span. Agreement of shock strength and position is very good. At $0.515 C_L$, agreement is equally good except near the tip where the experimental shock has shifted forward (fig. 5). Oil flow photographs indicate that there is probably some trailing-edge separation outboard of 90% semi-span at this lift. Shock induced separation has spread to mid-span at the next higher lift coefficient 0.63 (fig. 6). It is interesting to note that small oscillations in experimental upper surface pressures appear to have a correspondence with theoretical oscillations.

FL028 wing-body predictions show much closer agreement with the experiment than do FL022 isolated wing predictions. Is this difference caused by body effects or by differences in the two codes? FL028 was run with an isolated wing in order to investigate this question. Isolated wing pressures are compared with test data for $0.34 C_L$ and $0.515 C_L$ in figures 7 and 8. Notice that pressure oscillations are no longer present. Both FL028 and FL022 predict similar pressure distribution for an isolated wing although FL028 predicts a slightly stronger shock. It appears that the improved experiment-theory correlation observed with FL028 is due to the inclusion of body effects rather than code differences.

FL030 COMPARISON WITH EXPERIMENT

FL030 is a finite volume wing-body transonic analysis code developed by Caughey (ref. 2). This code uses body oriented coordinates instead of the Joukowski transformation of FL028. Geometry input for the 12% wing was identical to FL028 input except the nose was not closed and the wing was raised back to its original high position on the fuselage (fig. 9). A 9% thick supercritical wing was also investigated.

No geometry problems were encountered in running FL030. Most analyses were run 200 iterations on a crude grid and 200 iterations on a moderate grid of $80 \times 12 \times 16$ points. One solution was run an additional 200 iterations on a fine grid, $160 \times 24 \times 32$. The fine grid solution showed very little pressure change over the moderate grid solution except at the leading edge (fig. 10) yet required four

times the computer time (3720 sec vs 950 sec). A pressure spike approximately two chord lengths aft of the wing root trailing edge was evident on all runs. While it is not clear that this anomaly degraded pressures on the wing, there is an obvious error in the code.

Theory and experiment are compared for the 9% thick wing-body configuration in figures 11 through 13 for $0.28 C_L$, $0.416 C_L$, and $0.57 C_L$. Agreement is generally good prior to separation except near the tip. Model manufacturing errors may account for pressure differences at the tip and at the leading-edge near mid-span. The effect of a 0.01 Mach number change on FL030 wing pressures is shown in figure 14. The differences are concentrated in the vicinity of the shock and are largest near the tip.

Comparison of theory with experiment for the 12% thick wing is presented in figure 15 for the design condition. Agreement is good, with the largest errors being in the predicted shock strength at 0.4 semi-span and shock position at 0.88 semi-span. Results at $0.52 C_L$ are shown in figure 16. As discussed earlier, some trailing-edge separation exists for this condition at the wing tip.

Although both FL028 and FL030 show good agreement with the experiment, significant differences exist between predicted pressures (fig. 17). The wing input geometry was identical for both codes. FL030 was easier to use for the configurations investigated and did not require angle-of-attack adjustments to match experimental lift.

WIBCO COMPARISON WITH EXPERIMENT

WIBCO is a small perturbation transonic wing-body analysis code written by Boppe (ref. 4). A two-dimensional boundary layer subroutine is linked to this code which adds a displacement thickness to the wing geometry and updates it at specified intervals. The 12% thick supercritical wing was analyzed using WIBCO on NASA Langley's CYBER 175 computer. In order to save time, the wing was run on a circular fuselage in a mid-wing position.

Predicted pressures are compared with the experiment at $0.34 C_L$ and $0.52 C_L$ in figures 18 and 19. Theory was run at the test angles-of-attack. Agreement is surprisingly good considering the geometric and code approximations.

Isolated wing and wing-body predictions are compared in figure 20. Relatively little difference is seen between isolated wing and wing-body predicted pressures except at the root. Significant differences were predicted by FL028 across the entire span.

CONCLUSIONS

Three wing-body analysis codes have been evaluated by comparing predicted wing pressure distributions with experiment. All three codes show significant improvement over the isolated wing code, FL022. FL028 was very difficult to use for the particular fuselage analyzed. FL030 was much easier to use than FL028 but was more expensive. The simplest method evaluated, WIBCO, gave very good results without excessive

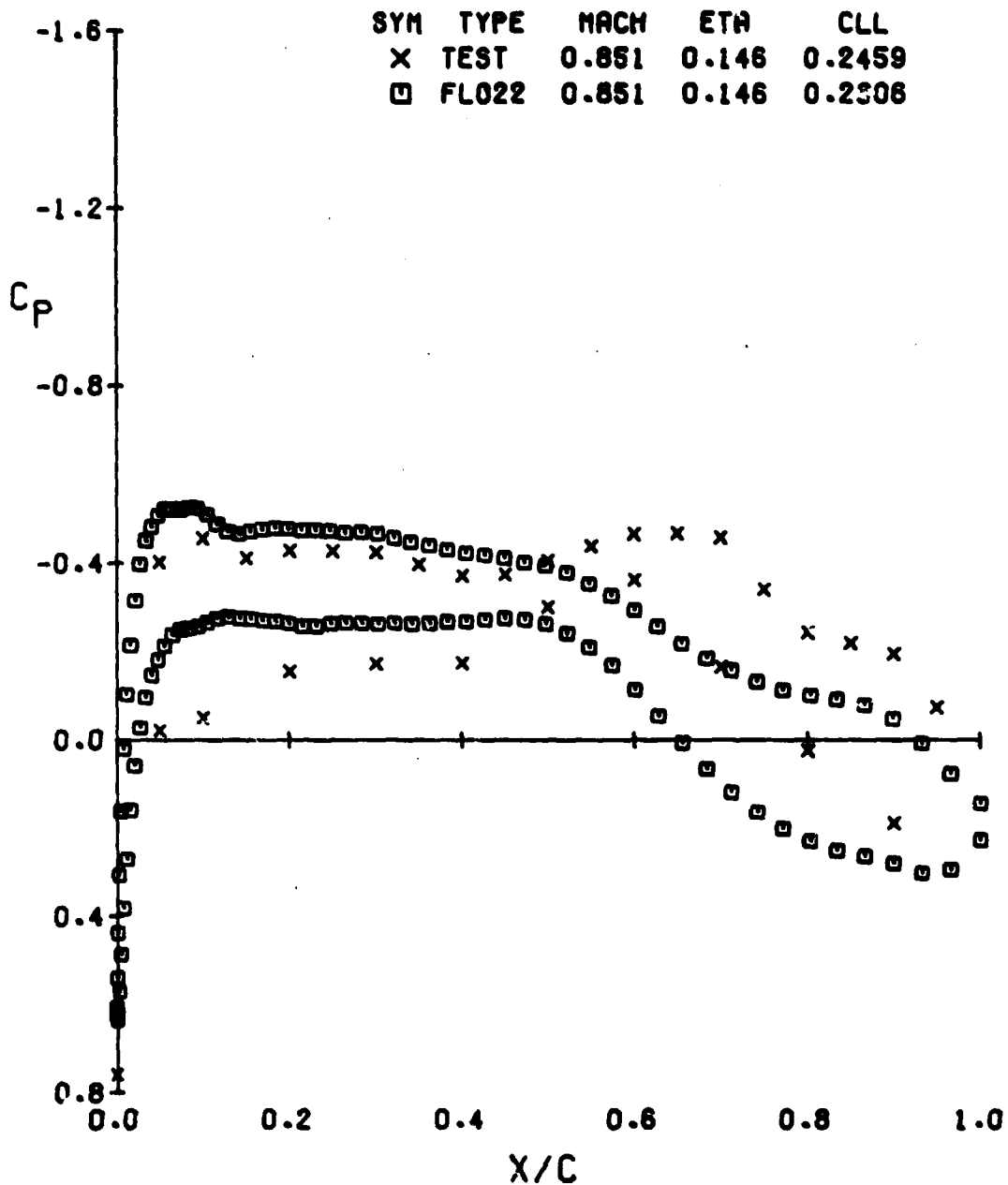
run times. Unfortunately, all codes investigated are too expensive to be used for extensive design perturbations.

ACKNOWLEDGMENT

Contract monitor for this study was Mr. Paul Scheurich, DTNSRDC. Dr. Jerry South, Theoretical Aerodynamics Branch of NASA Langley Research Center, made computer time available for WIBCO analyses. Mr. Ed Waggoner assisted in running the WIBCO code.

1. Jameson, Antony; and Caughey, D. A.: Numerical Calculation of the Flow Past a Swept Wing. ERDA Report COO-3077-140, June 1977.
2. Caughey, D. A.; and Jameson, A.: Recent Progress in Finite-Volume Calculations for Wing-Fuselage Combinations. AIAA Paper 79-2523, 23th Fluid and Plasma Dynamics Conference, July 1979.
3. Haney, H. P.; and Johnson, R. R.: Application of Numerical Optimization to the Design of Wings with Specified Pressure Distributions. NASA CR-3238, February 1980.
4. Boppe, C. W.: Calculation of Transonic Wing Flows by Grid Embedding. AIAA Paper 77-207, January 1977.

ISOLATED WING NO. 1



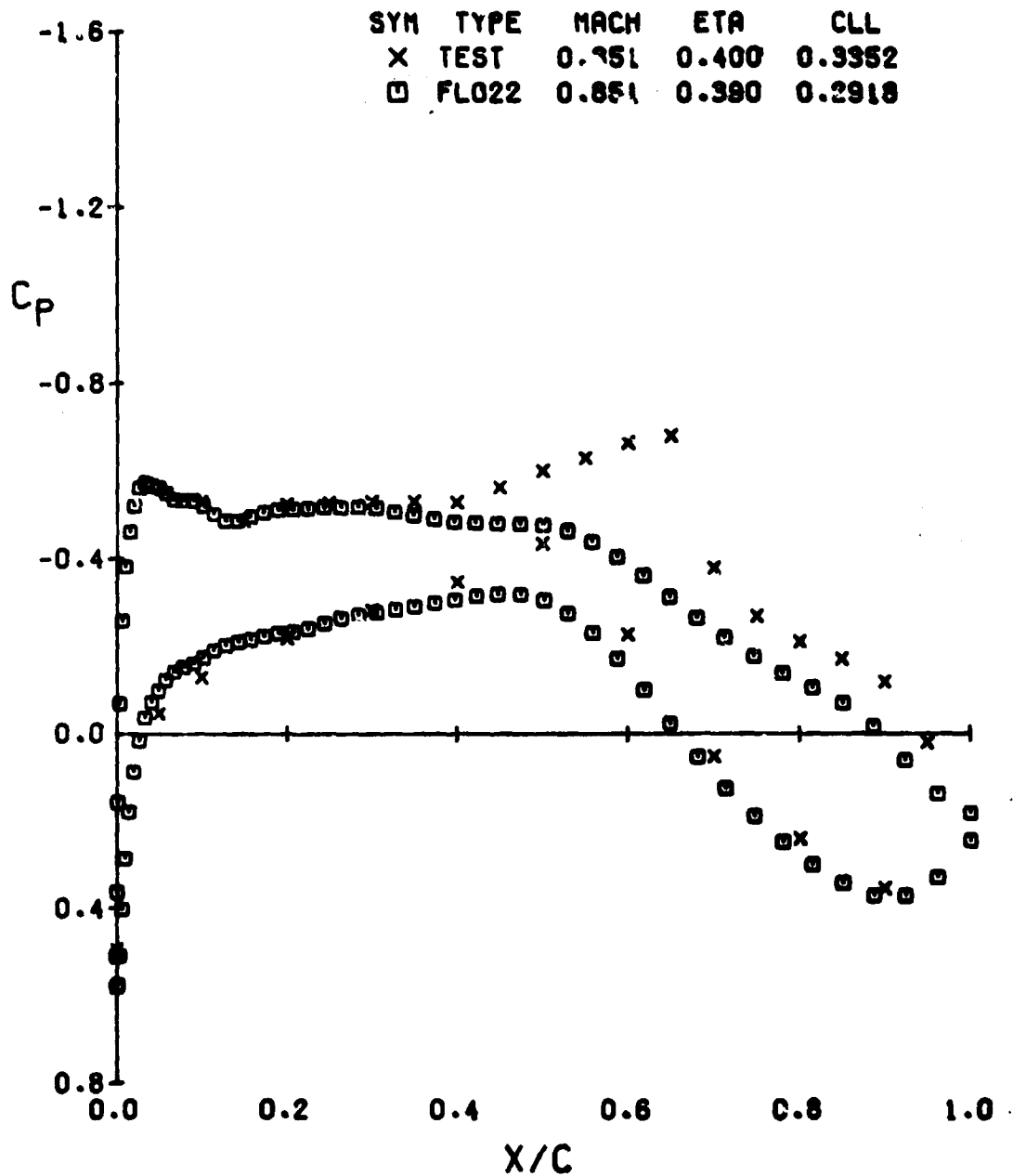
(A) $\text{ETA} = 0.146$

FIGURE 1.- COMPARISON OF FLO22 WING PRESSURES WITH EXPERIMENT

$\text{ALPHA} = 4.68$

$\text{CL} = .34$

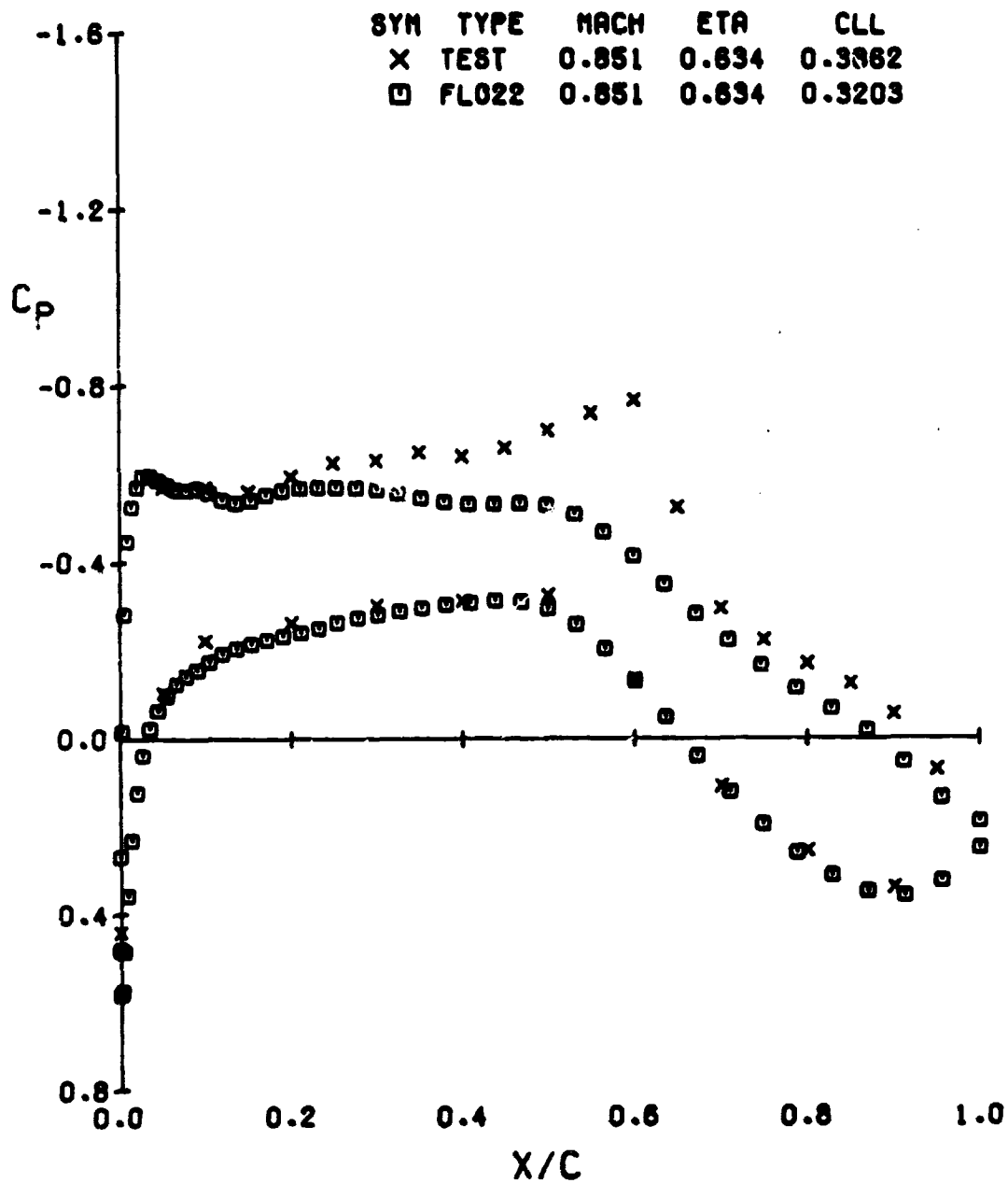
ISOLATED WING NO. 1



(B) $\eta = 0.400$

FIGURE 1.- CONTINUED

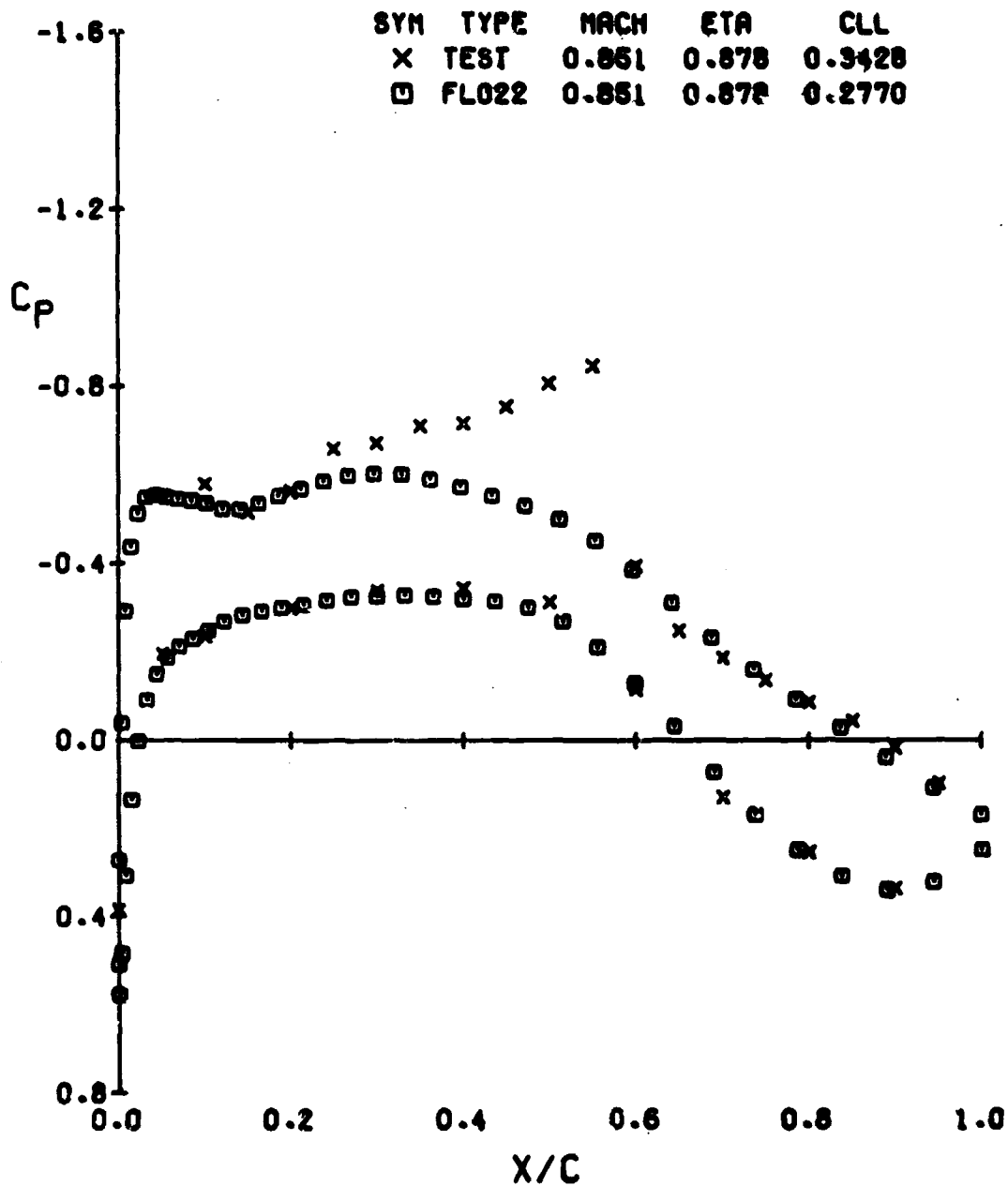
ISOLATED WING NO. 1



(C) $\text{ETA} \approx 0.634$

FIGURE 1.- CONTINUED

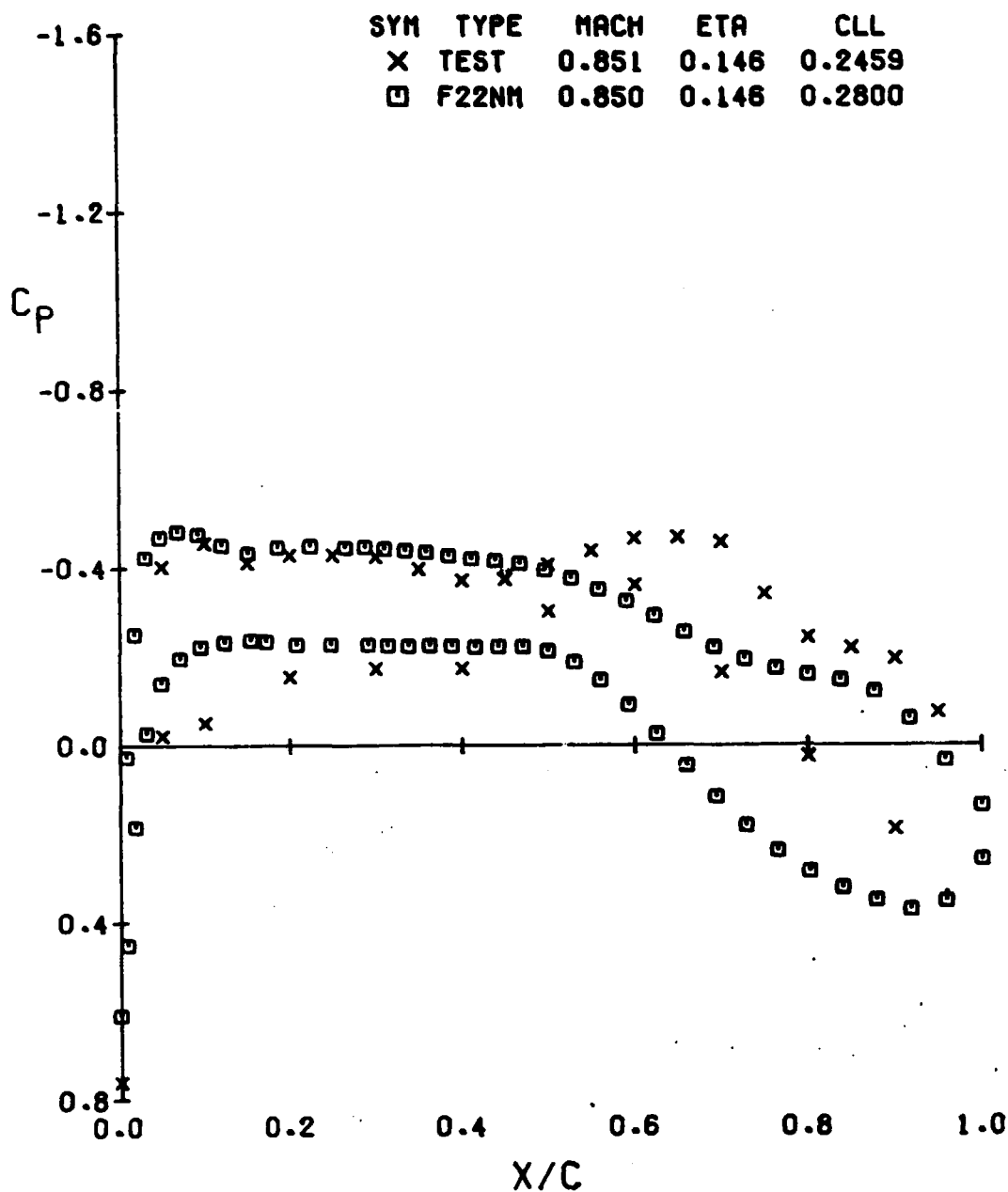
ISOLATED WING NO. 1



(D) $\text{ETA} = 0.878$

FIGURE 1.- CONCLUDED

ISOLATED WING NO. 1

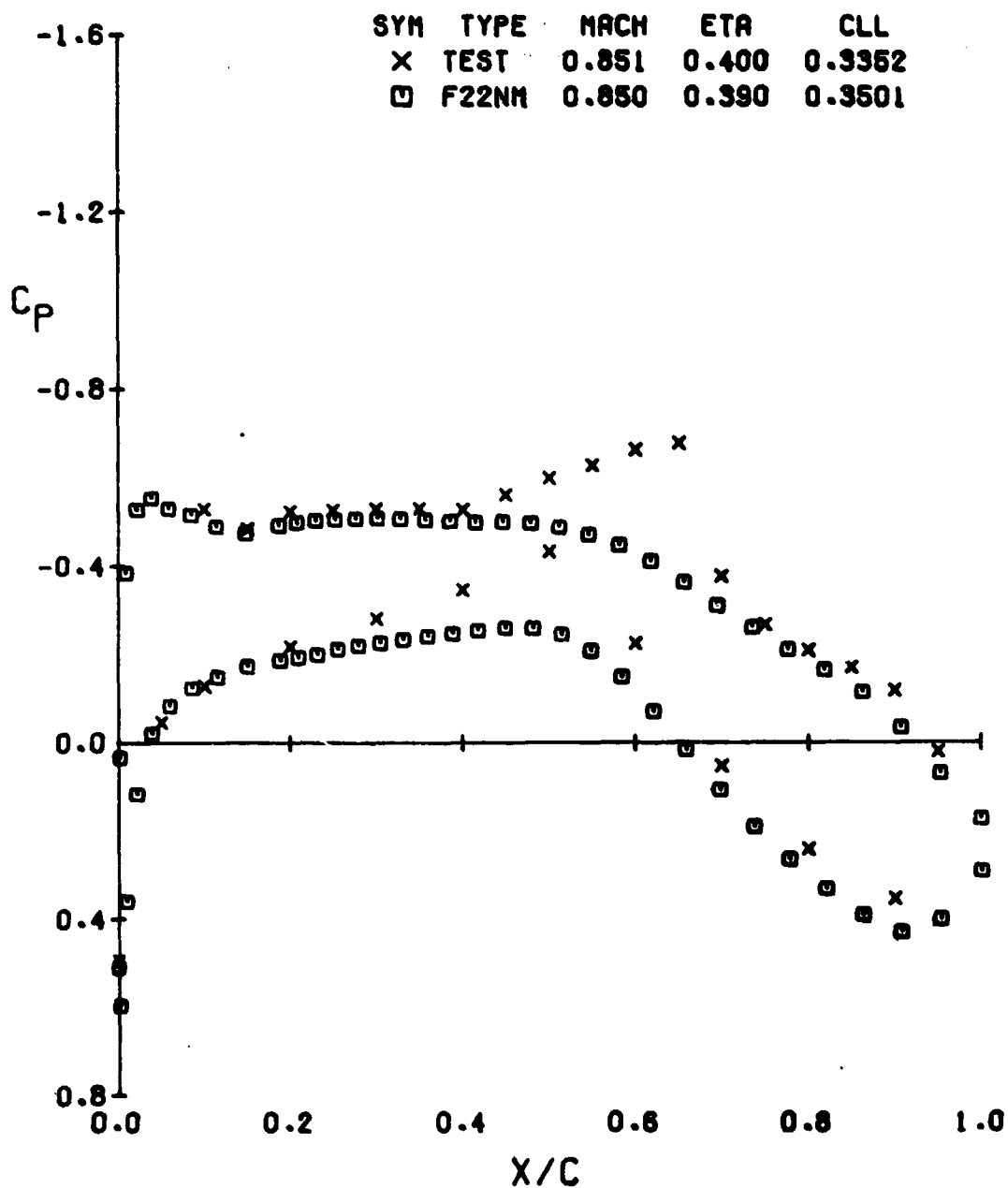


(A) $\eta = 0.145$

FIGURE 2.- COMPARISON OF FLO22NM WING PRESSURES WITH EXPERIMENT

$\alpha = 4.68$ $CL(EXP) = .34$

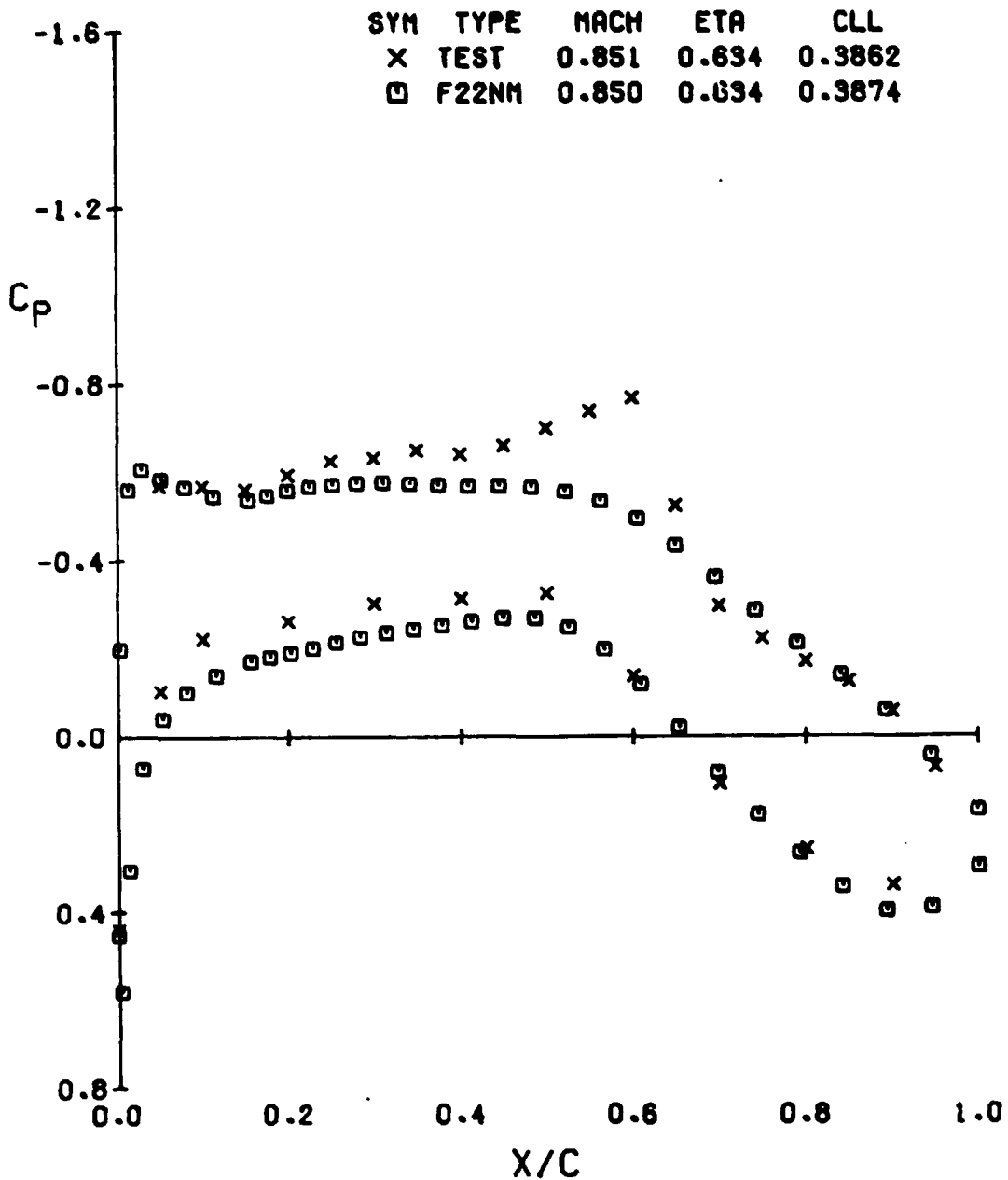
ISOLATED WING NO. 1



(8) $\text{ETA} = 0.400$

FIGURE 2.- CONTINUED

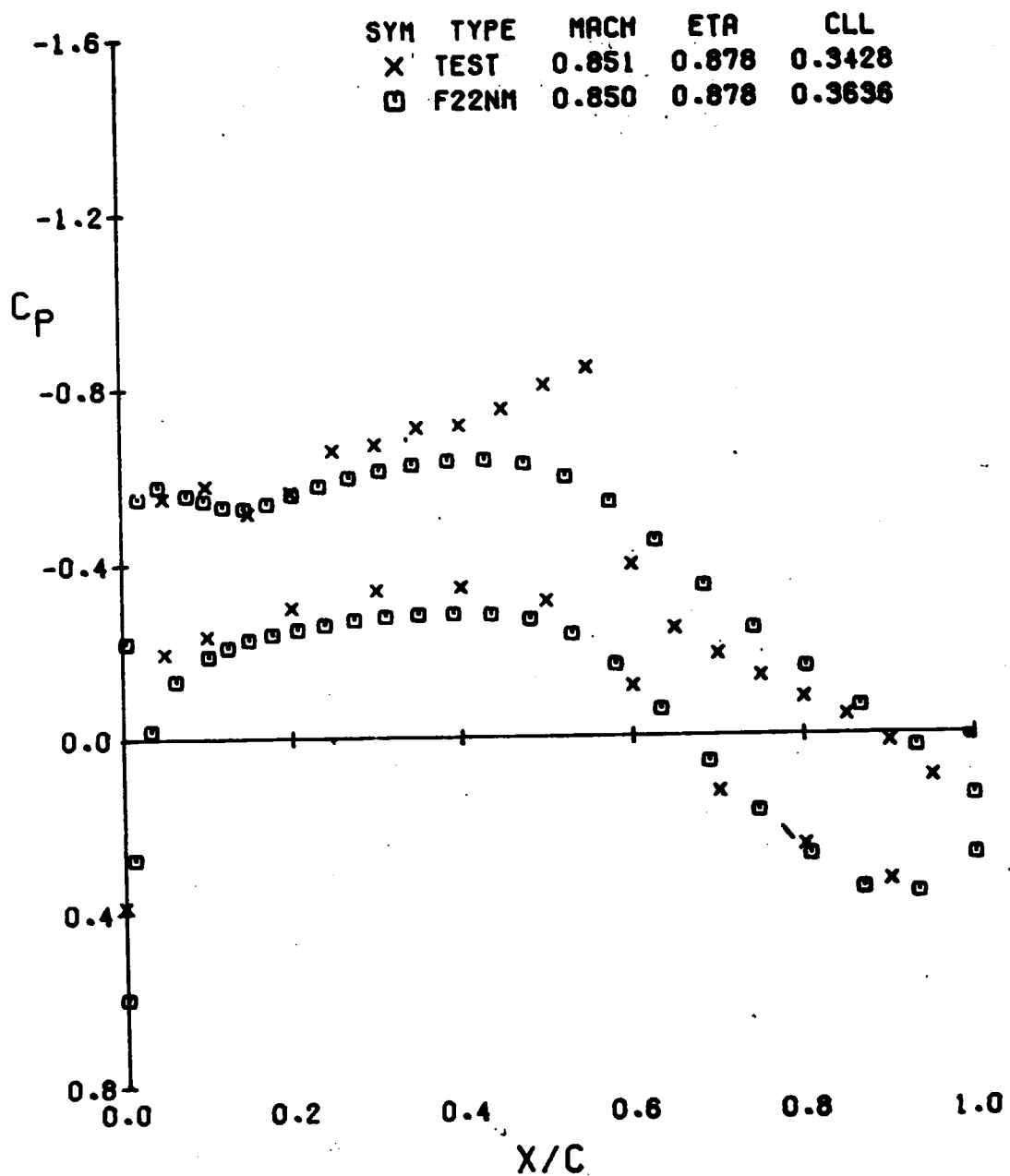
ISOLATED WING NO. 1



(C) $\text{ETA} = 0.634$

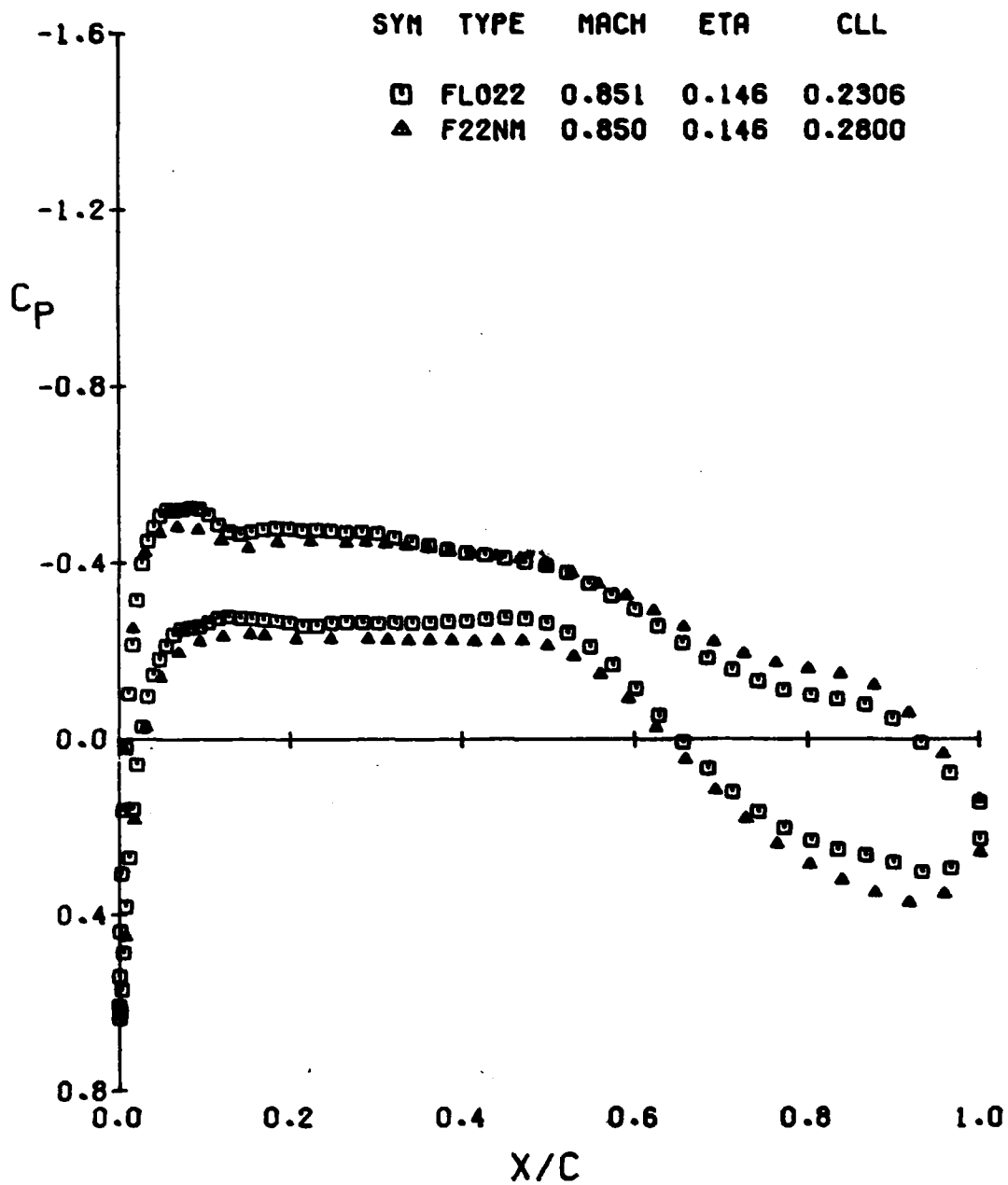
FIGURE 2.- CONTINUED

ISOLATED WING NO. 1



(D) $\text{ETA} = 0.878$
 FIGURE 2.- CONCLUDED

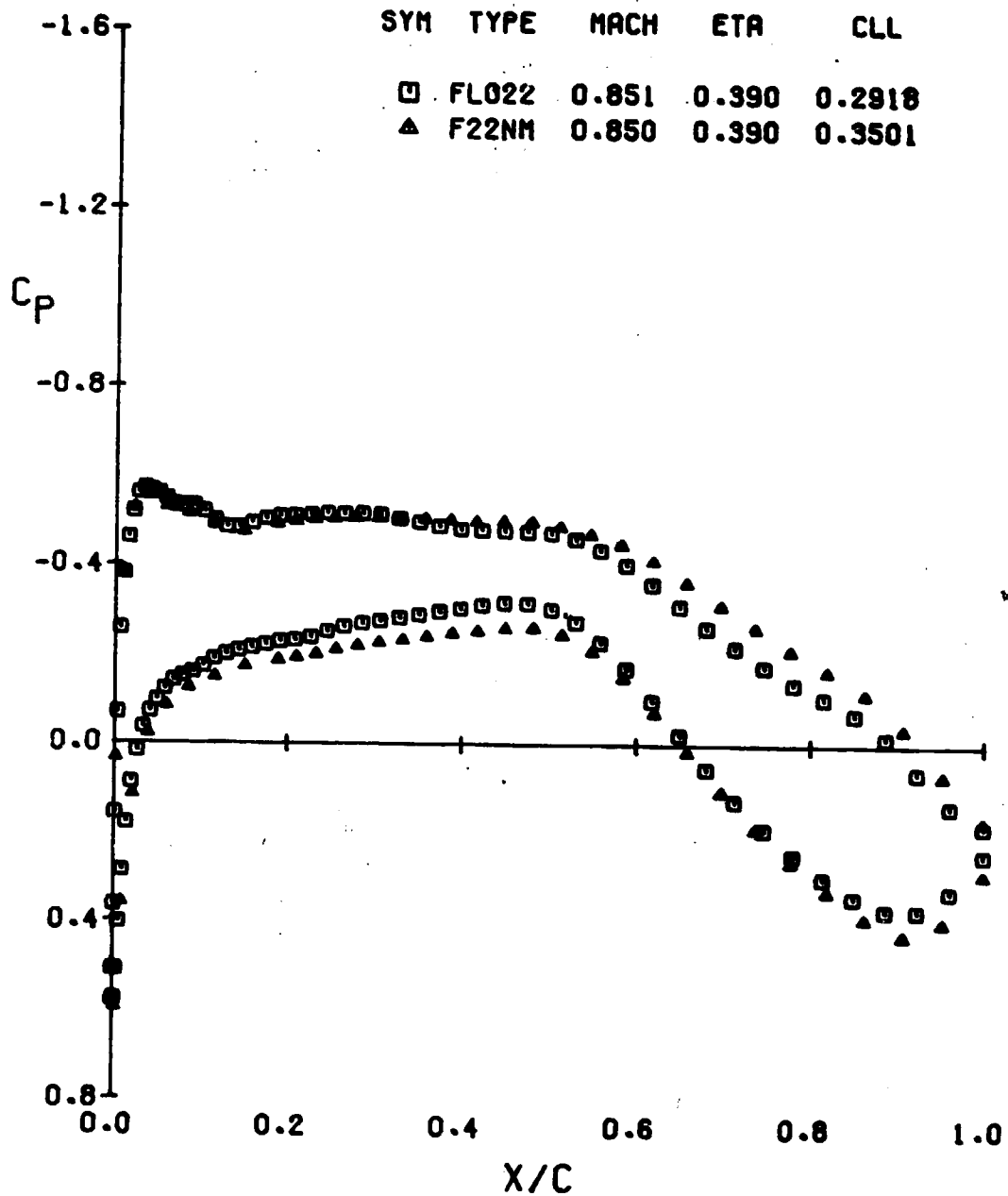
ISOLATED WING NO. 1



(A) $\text{ETA} = 0.146$

FIGURE 3.- COMPARISON OF FLO22 AND FLO22NM PRESSURE PREDICTIONS

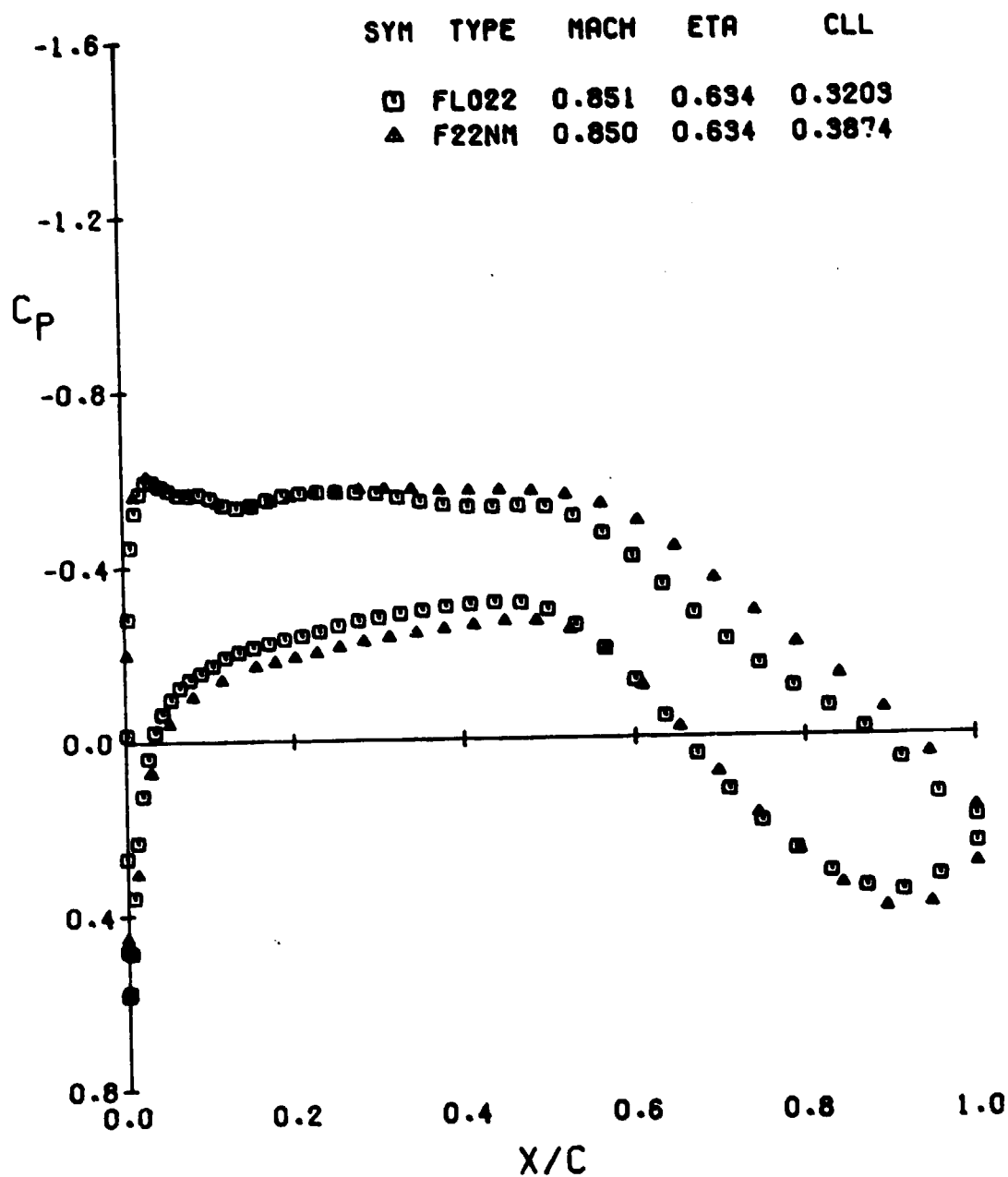
ISOLATED WING NO. 1



(B) $\text{ETA} = 0.390$

FIGURE 3.- CONTINUED

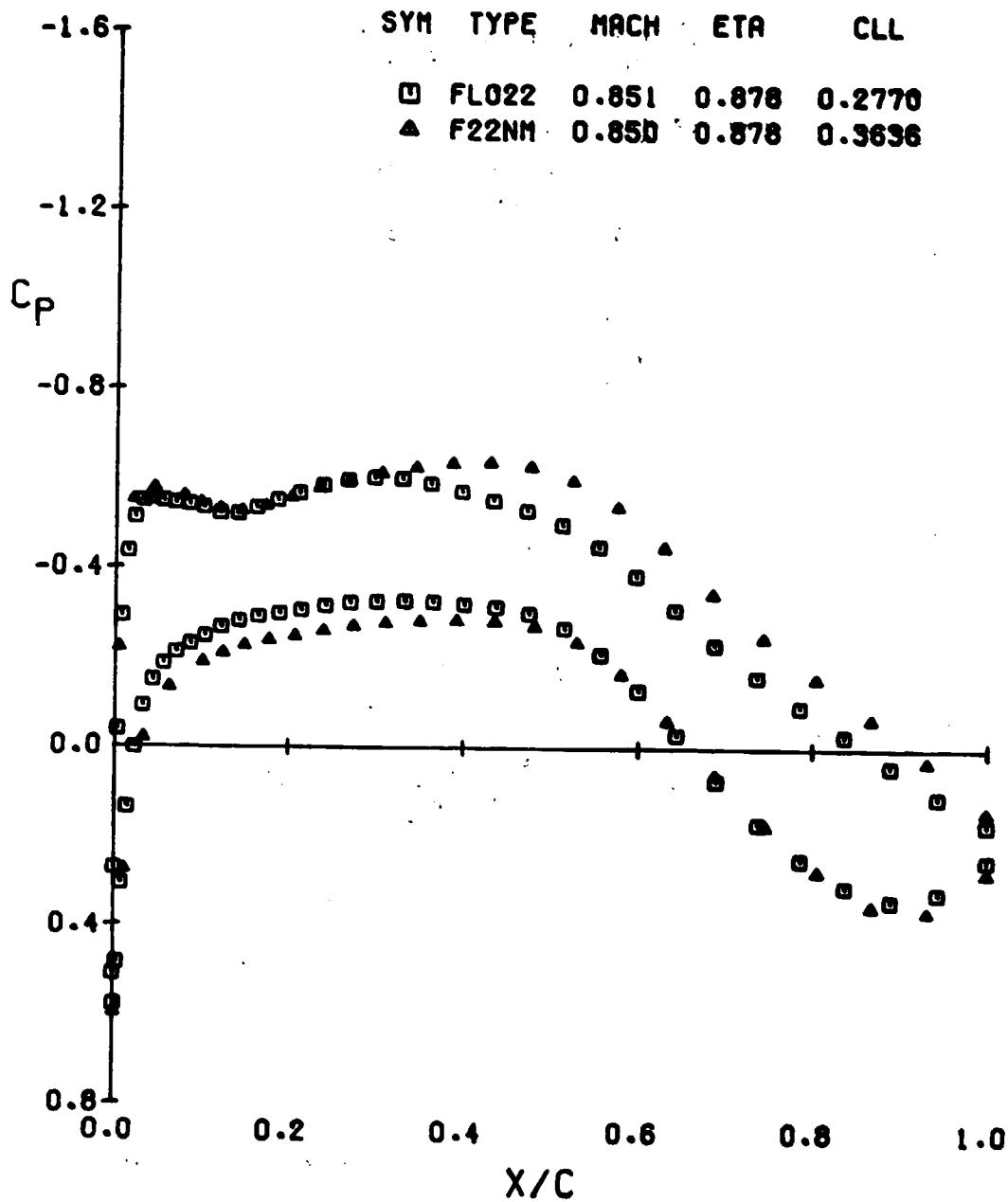
ISOLATED WING NO. 1



(C) $\eta = 0.634$

FIGURE 3.- CONTINUED

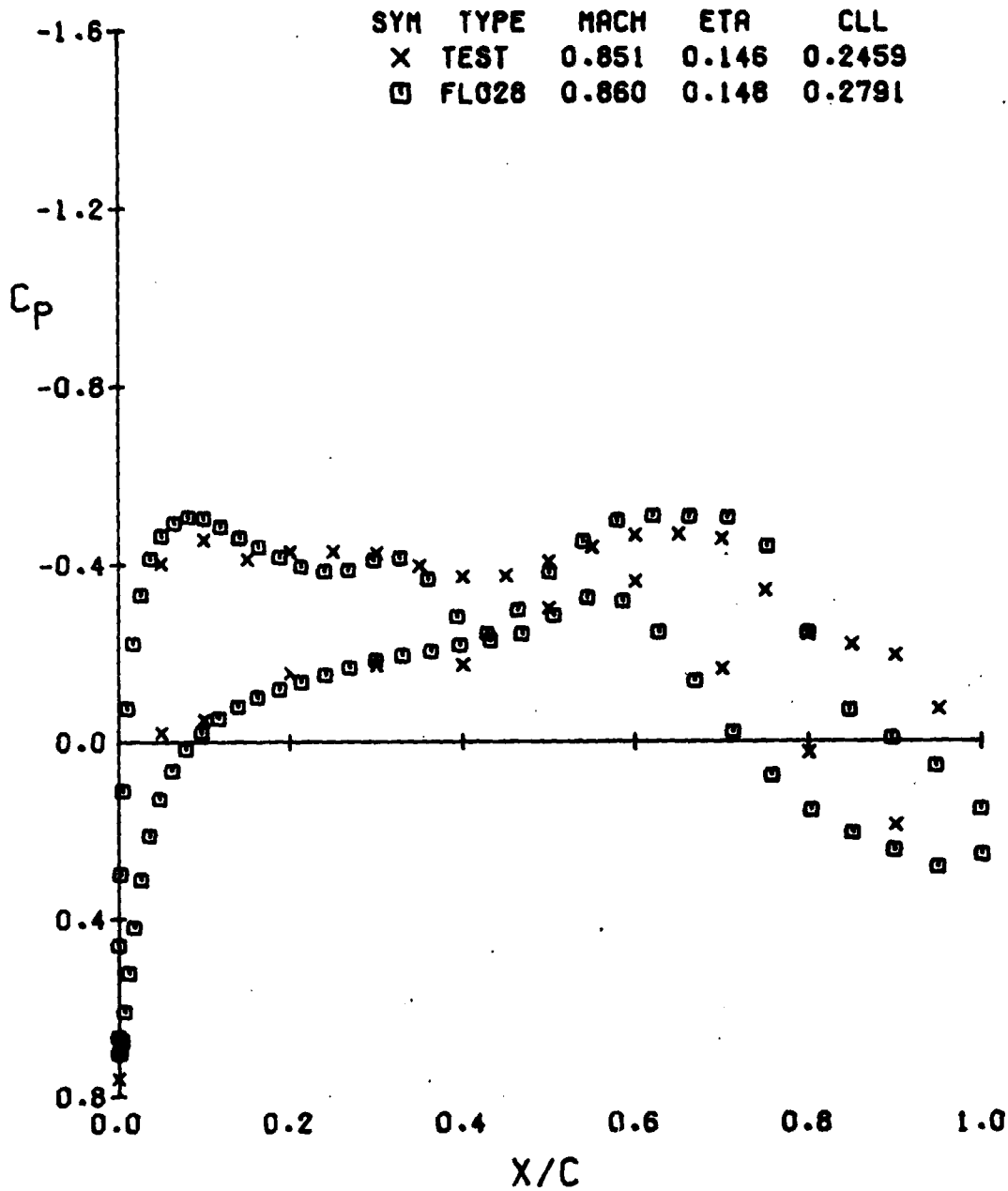
ISOLATED WING NO. 1



(D) ETA = 0.878

FIGURE 3.- CONCLUDED

BODY/WING NO.1 T/C=.12

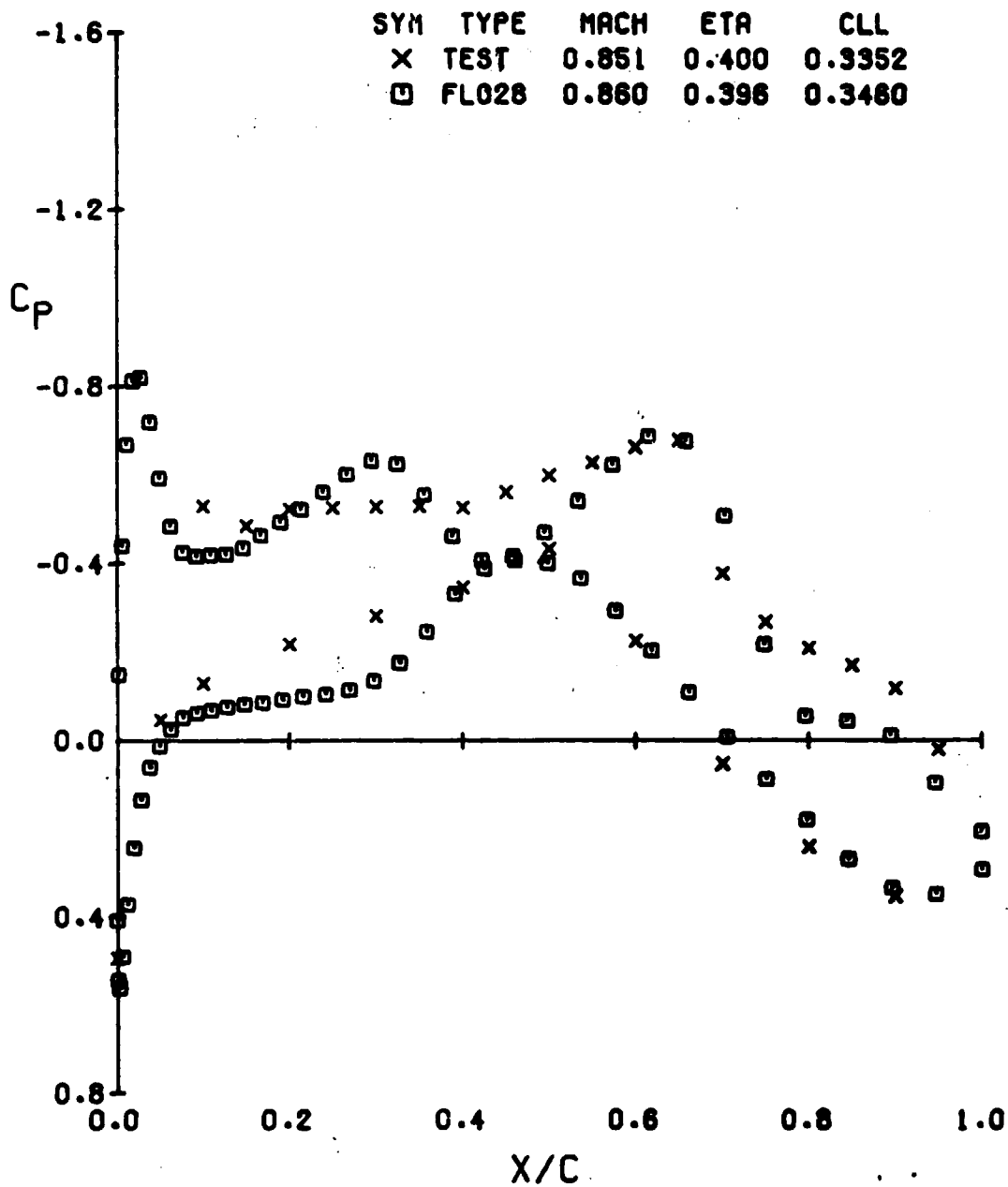


(A) $\text{ETA} = 0.146$

FIGURE 4.- COMPARISON OF FLO28 WING/BODY PRESSURES WITH EXPERIMENT

$\text{ALPHA} = 4.68$ $\text{CL(EXP)} = .34$

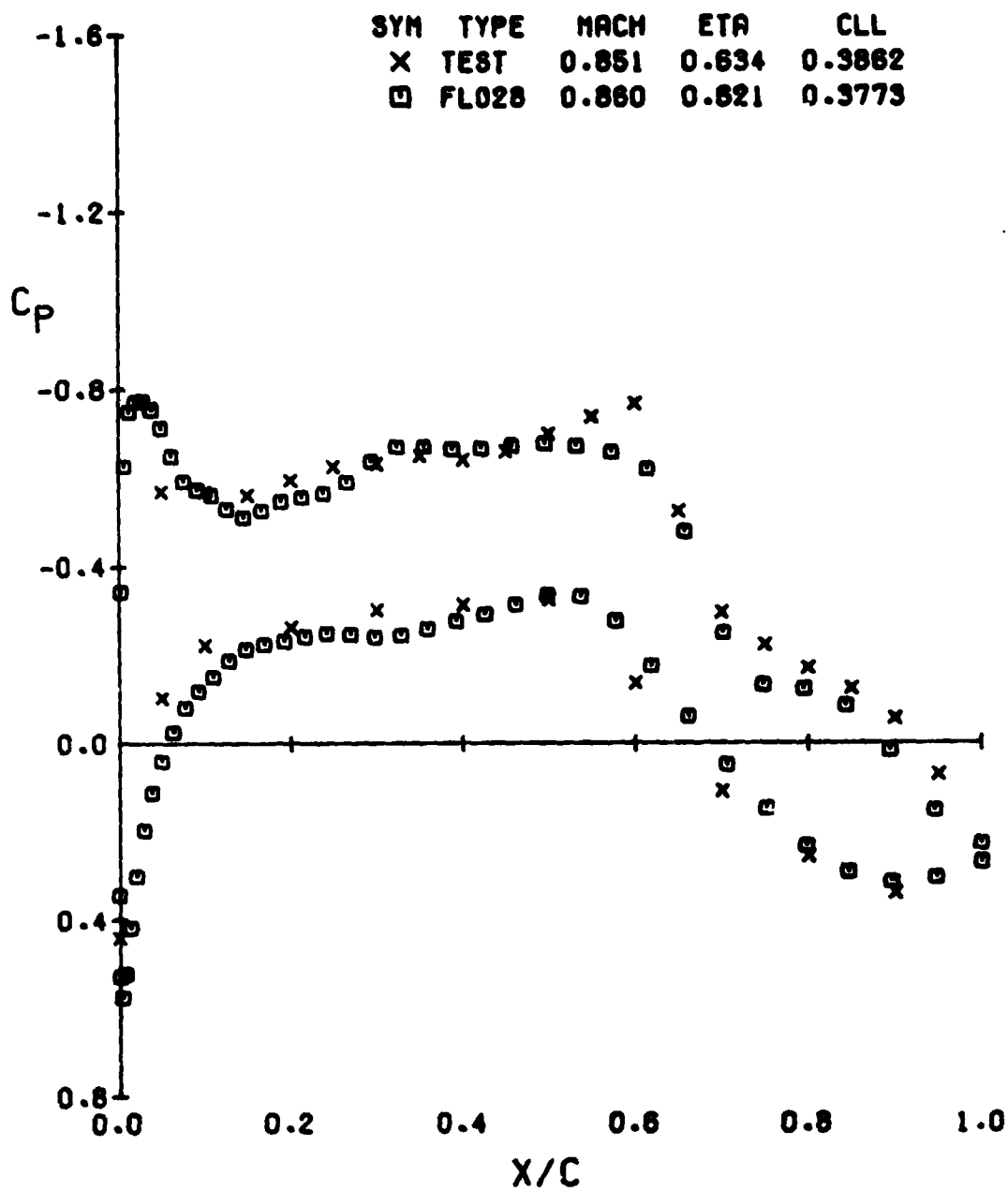
BODY/WING NO.1 T/C=.12



(B) ETA = 0.400

FIGURE 4.- CONTINUED

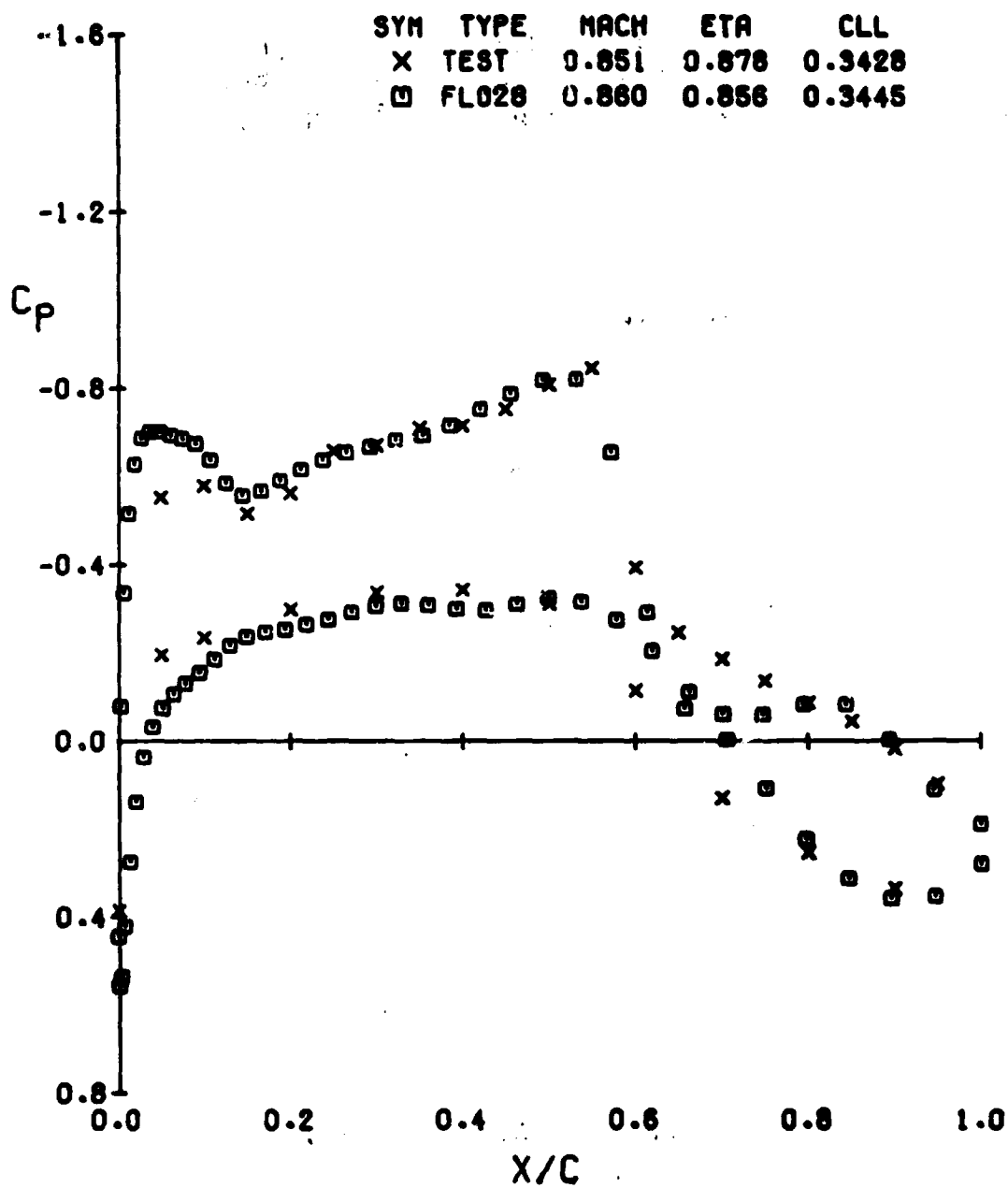
BODY/WING NO.1 T/C=.12



(C) ETA = 0.634

FIGURE 4.- CONTINUED

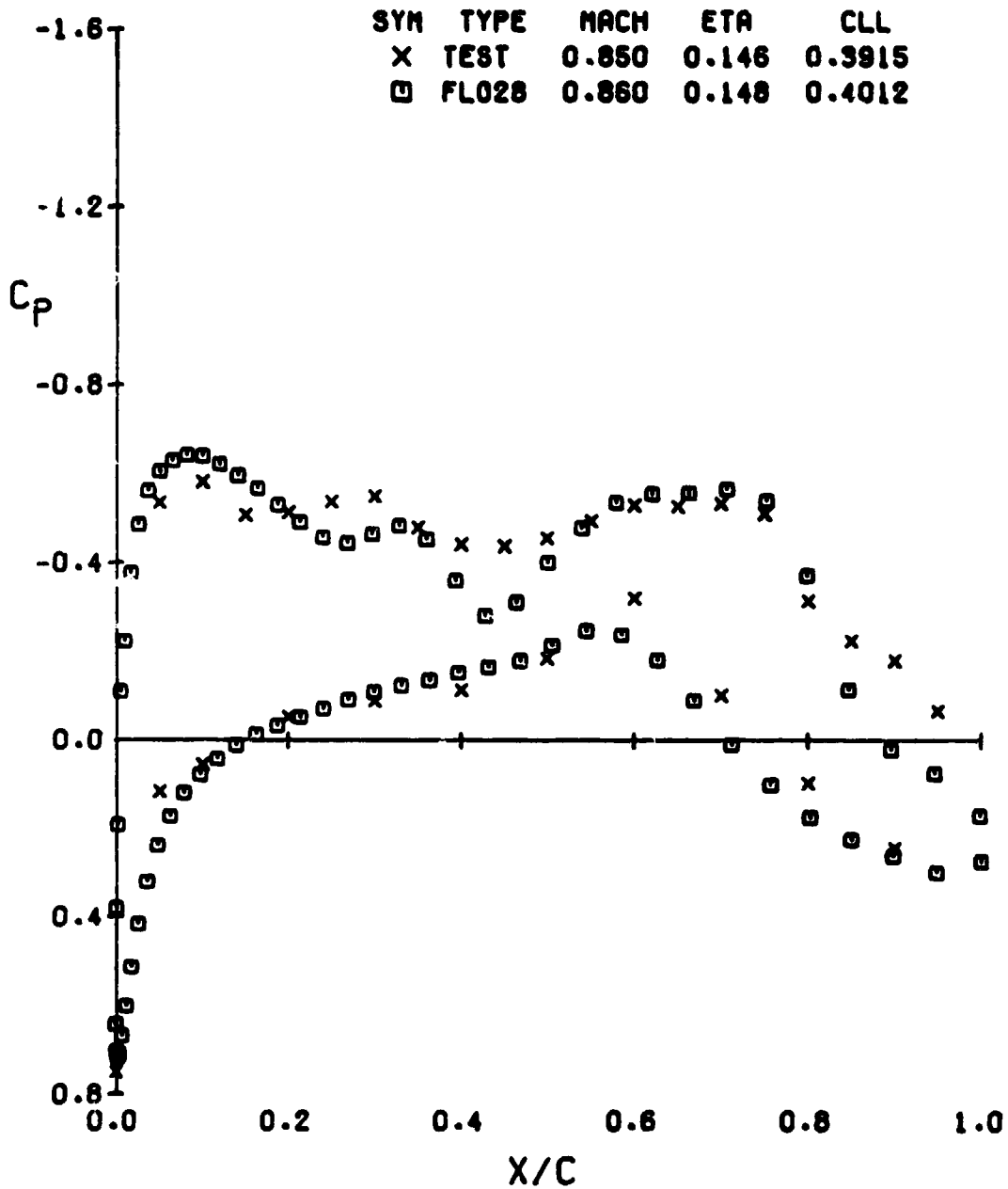
BODY/WING NO.1 T/C=.12



(D) ETA = 0.878

FIGURE 4.- CONCLUDED

BODY/WING NO.1 T/C=.12

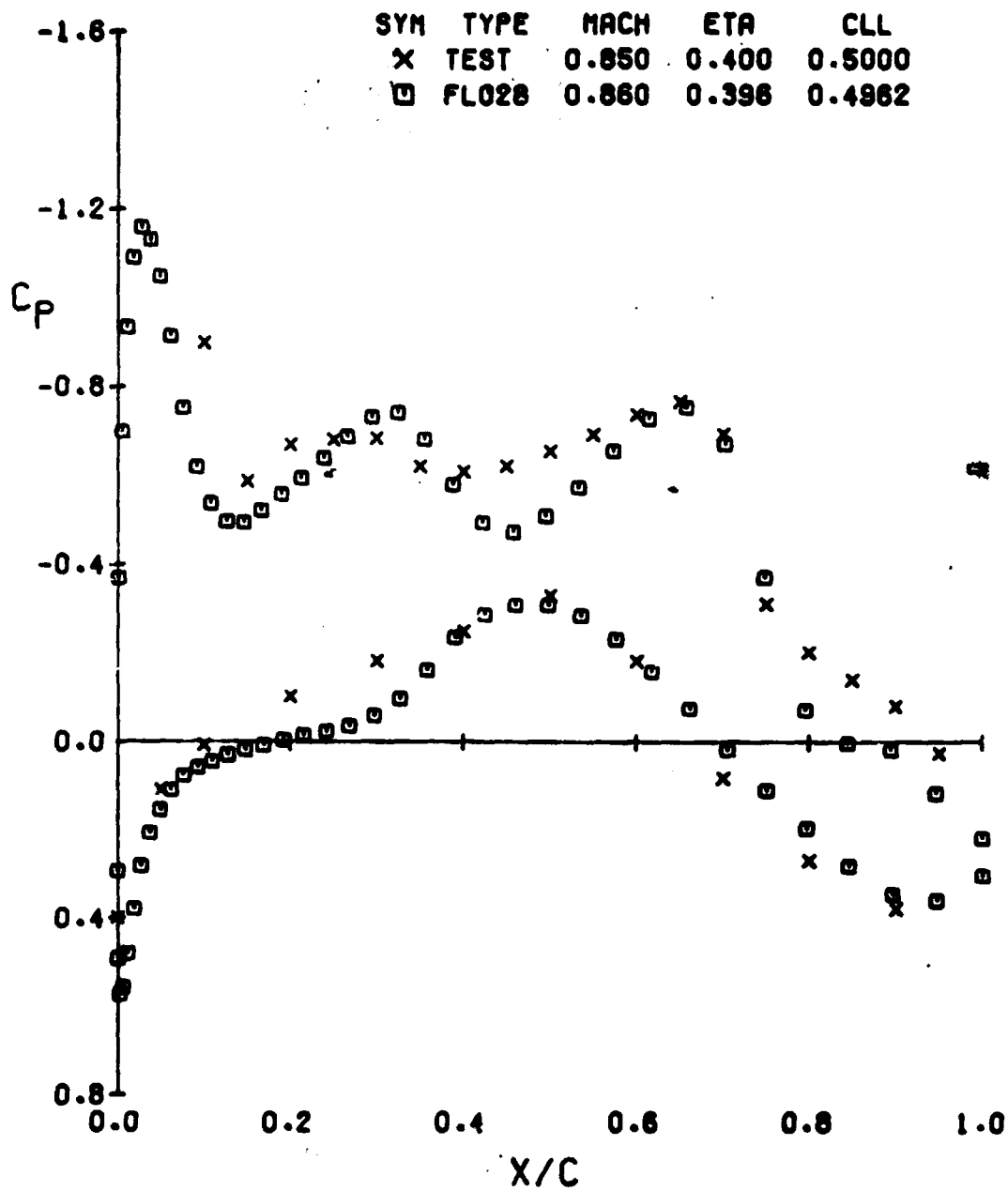


(A) ETA = 0.146

FIGURE 5.- COMPARISON OF FLO28 WING/BODY PRESSURES WITH EXPERIMENT

ALPHA = 6.52 CL(EXP) = .515

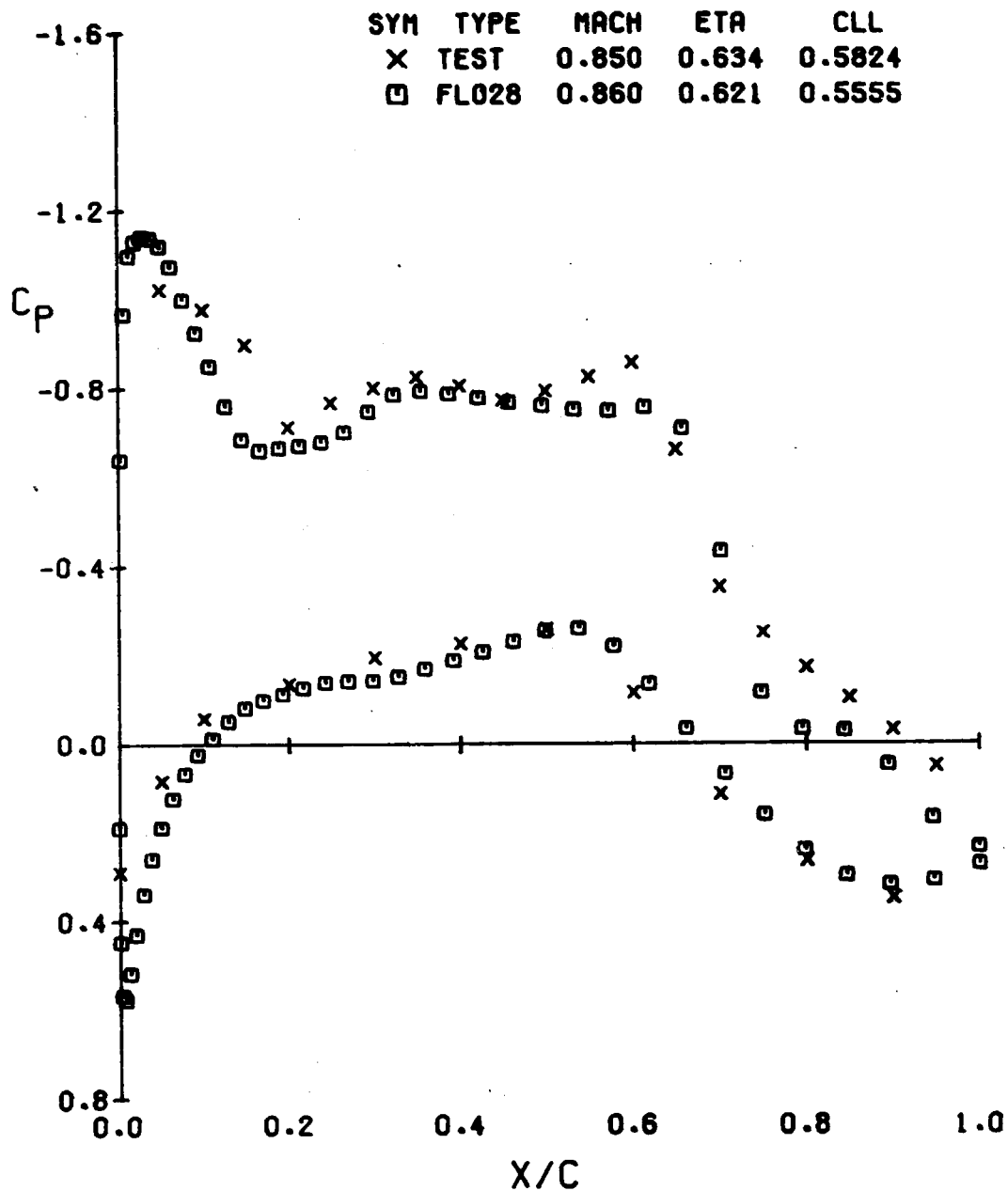
BODY/WING NO.1 T/C=.12



(B) ETA = 0.400

FIGURE 5.- CONTINUED

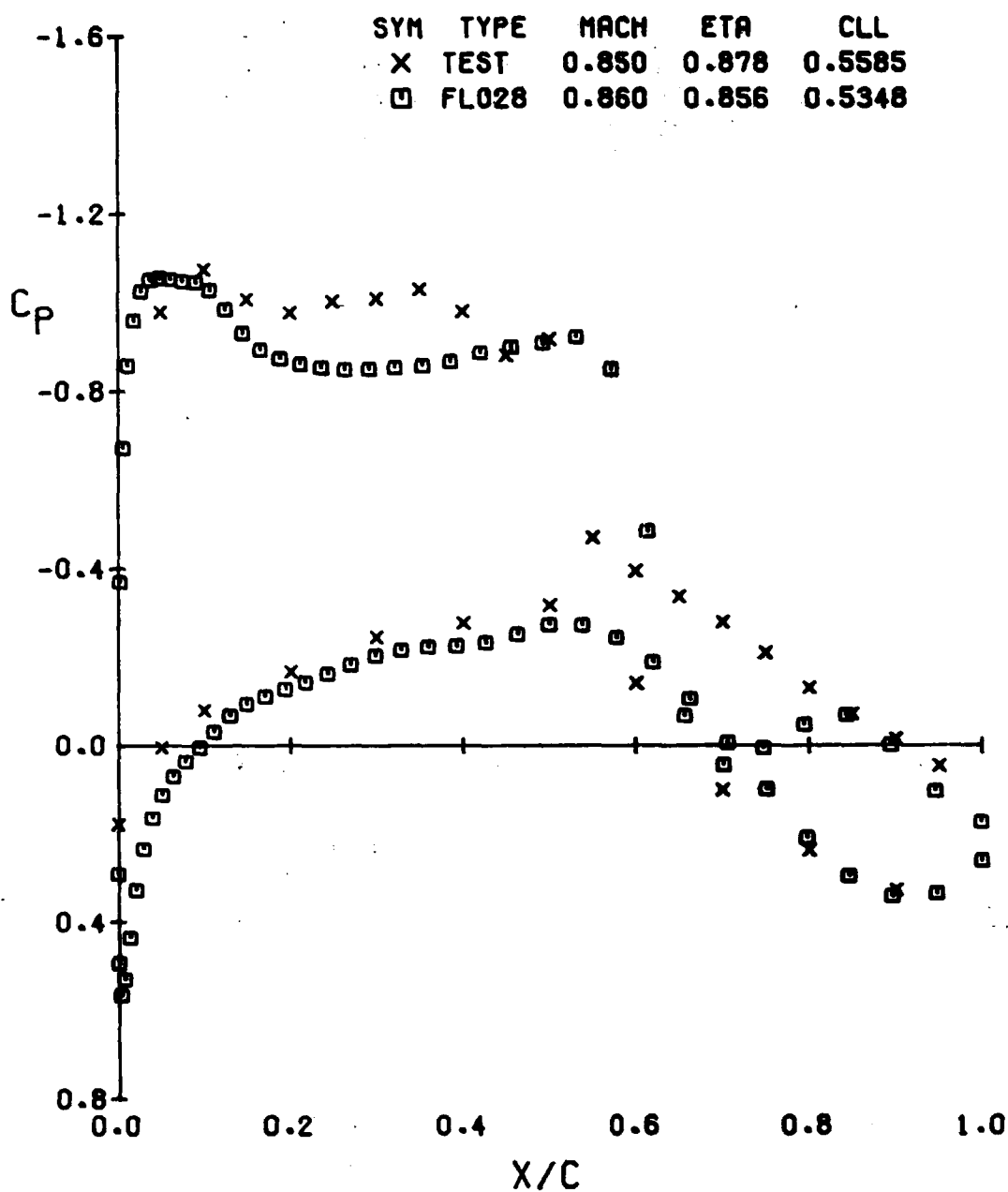
BODY/WING NO.1 T/C=.12



(C) ETA = 0.634

FIGURE 5.- CONTINUED

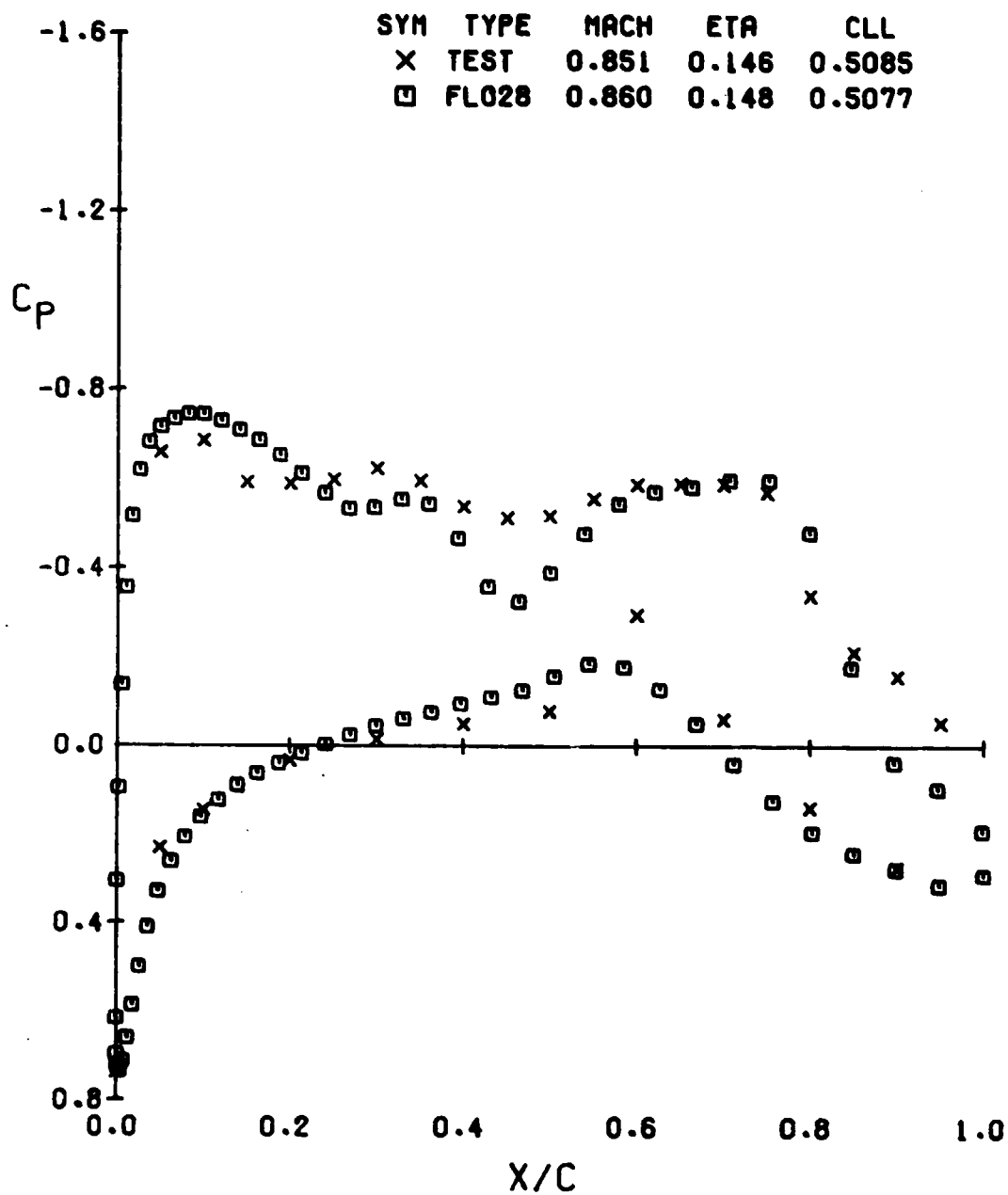
BODY/WING NO.1 T/C=.12



(O) ETA = 0.878

FIGURE 5.- CONCLUDED

BODY/WING NO.1 T/C=.12

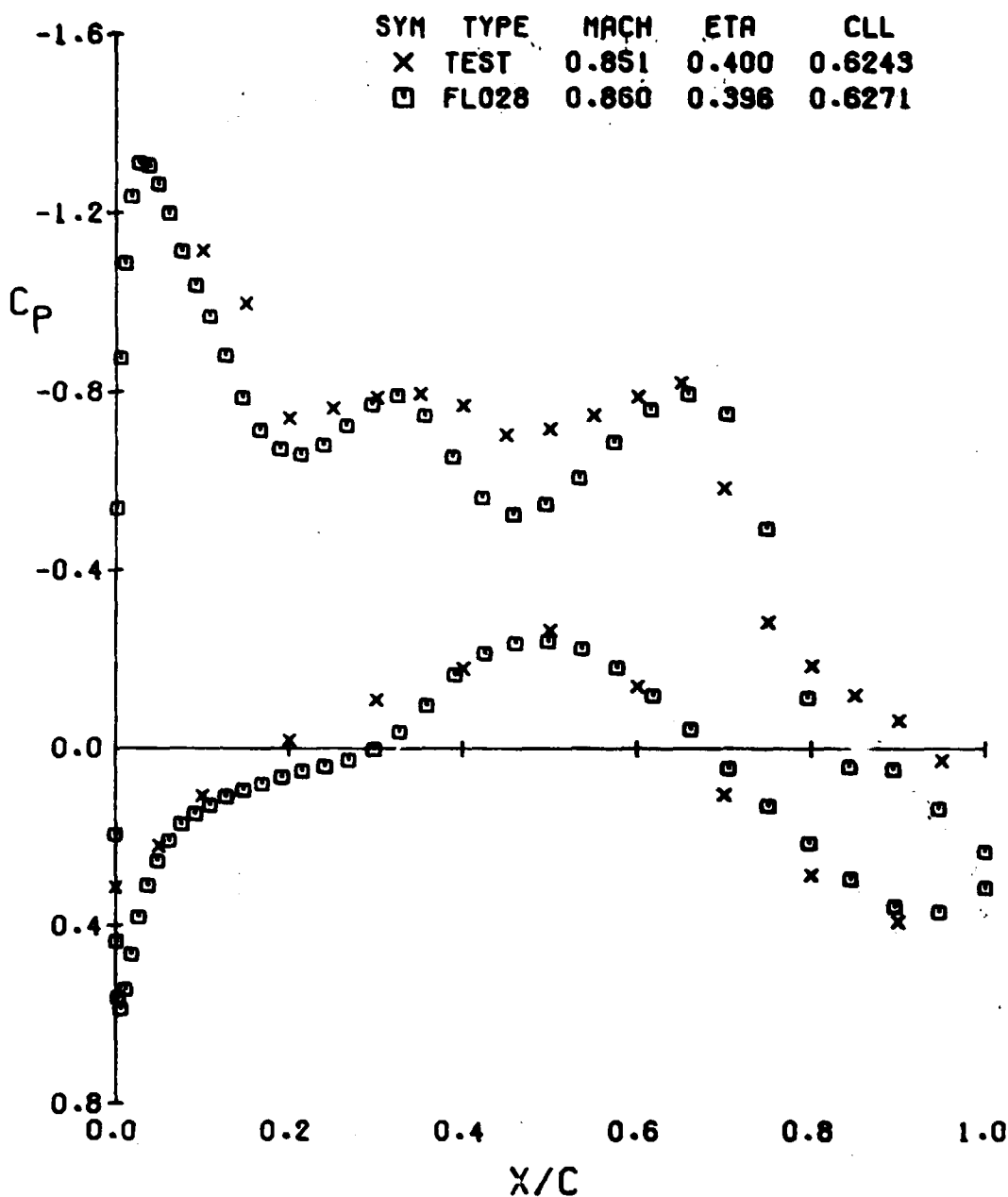


(A) ETA = 0.146

FIGURE 6.- COMPARISON OF FLO28 WING/BODY PRESSURES WITH EXPERIMENT

ALPHA = 8.14 CL(EXP) = .63

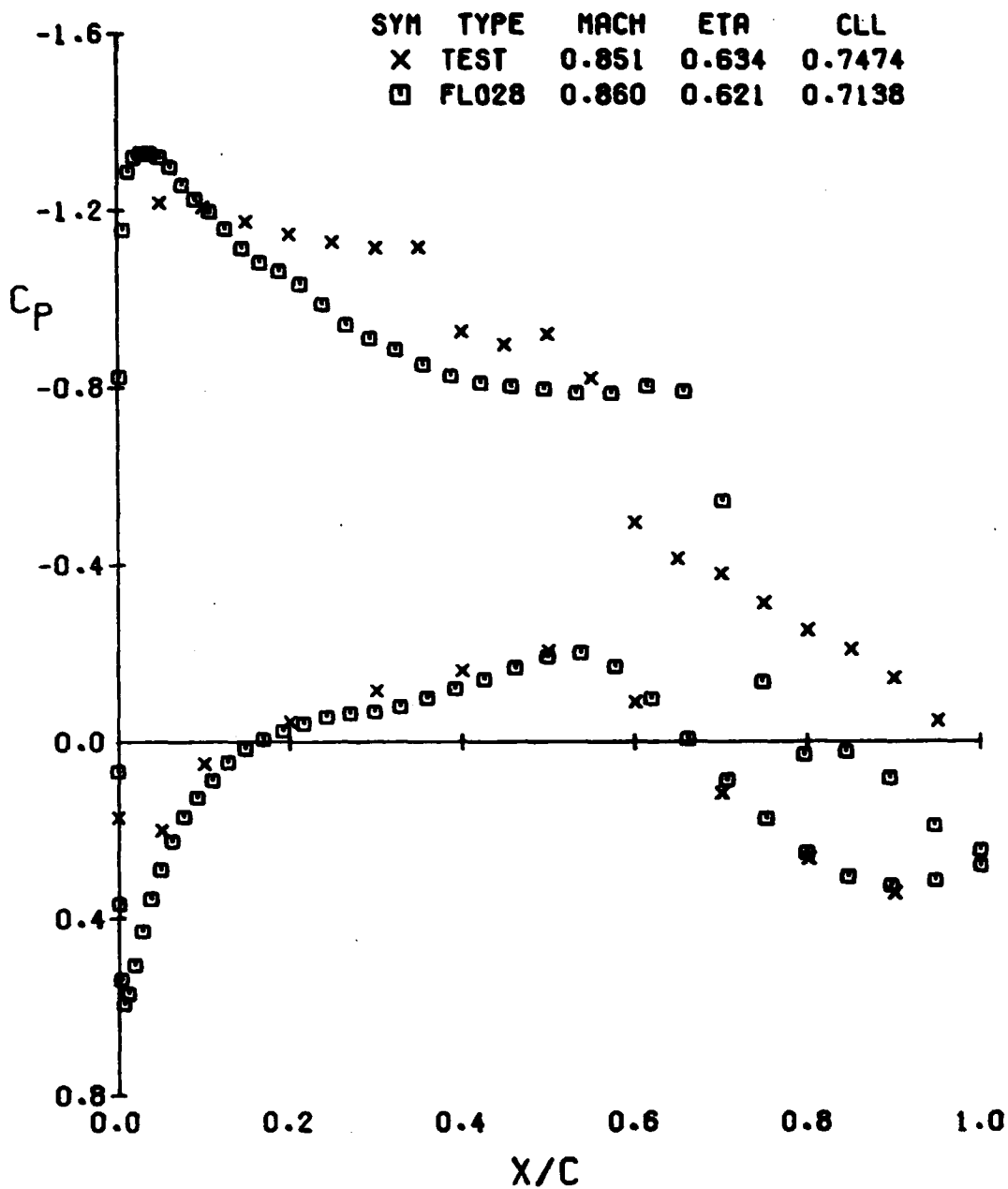
BODY/WING NO.1 T/C=.12



(B) ETA = 0.400

FIGURE 6.- CONTINUED

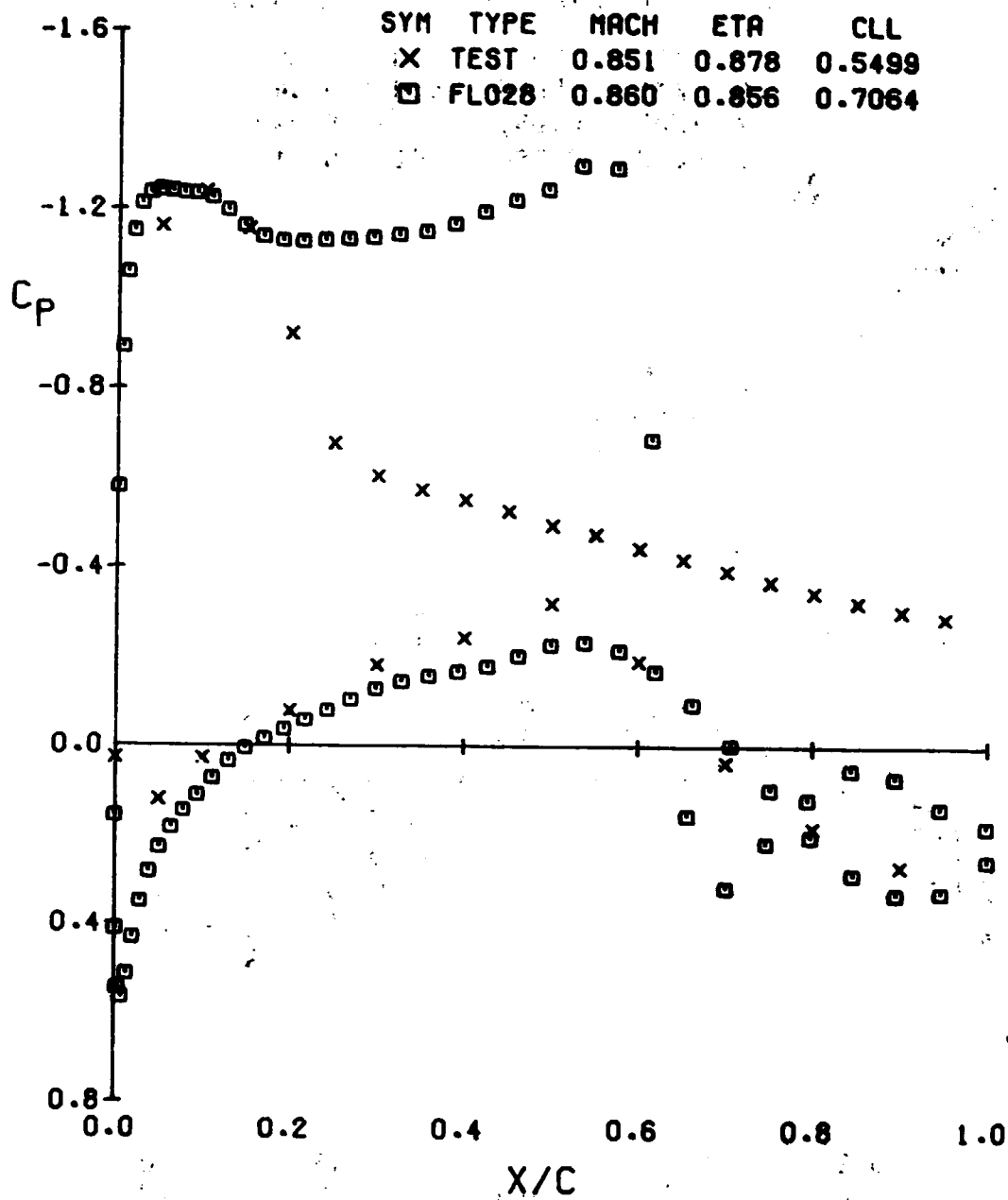
BODY/WING NO.1 T/C=.12



(C) $\text{ETA} = 0.634$

FIGURE 6.- CONTINUED

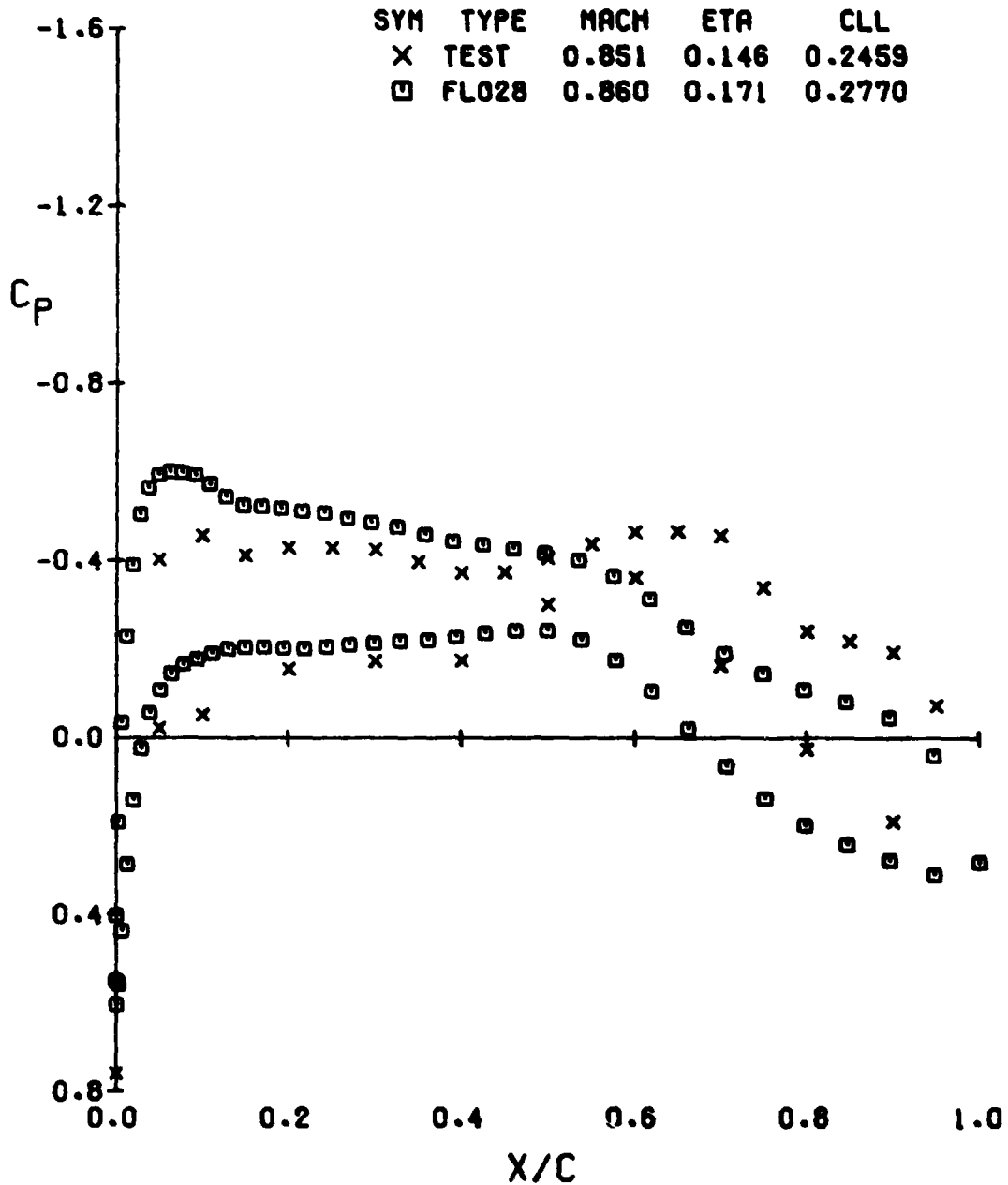
BODY/WING NO.1 T/C=.12



(D) ETA = 0.878

FIGURE 6. CONCLUDED

ISOLATED WING NO. 1

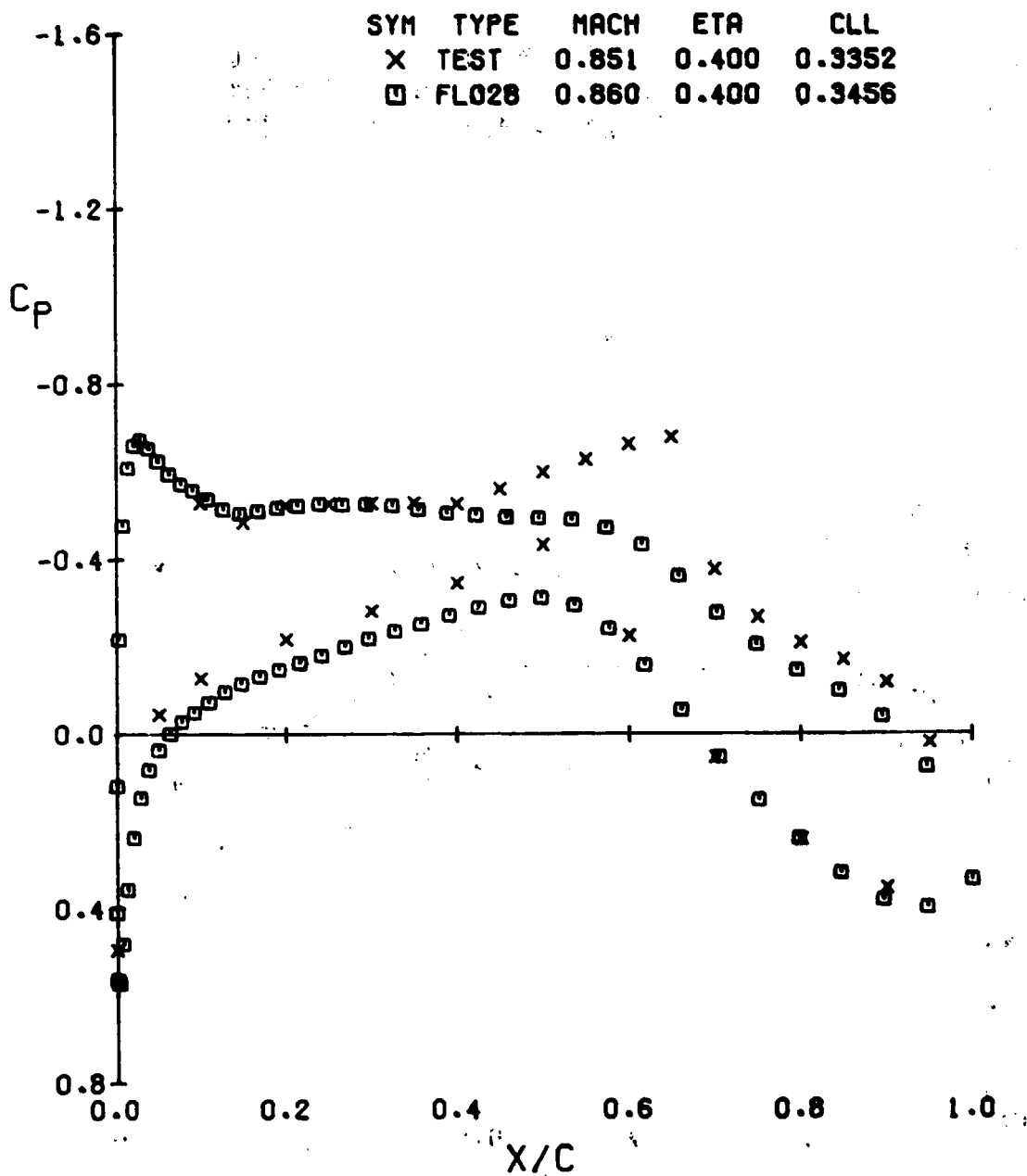


(A) $\eta = 0.146$

FIGURE 7.- COMPARISON OF FLO28 WING PRESSURES WITH EXPERIMENT

$\alpha = 4.68$ $CL(Exp) = .94$

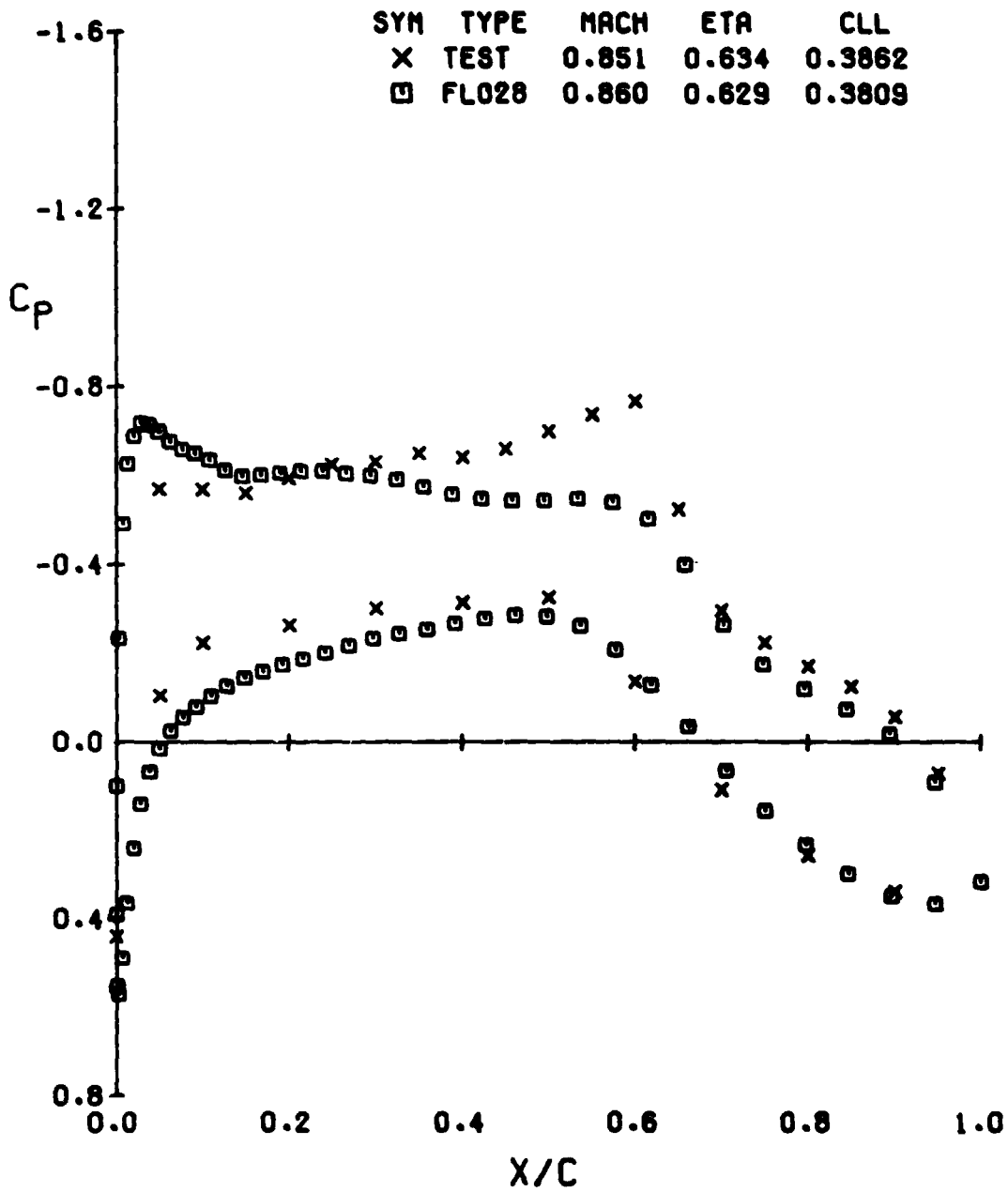
ISOLATED WING NO. 1



(B) ETA = 0.400

FIGURE 7.- CONTINUED

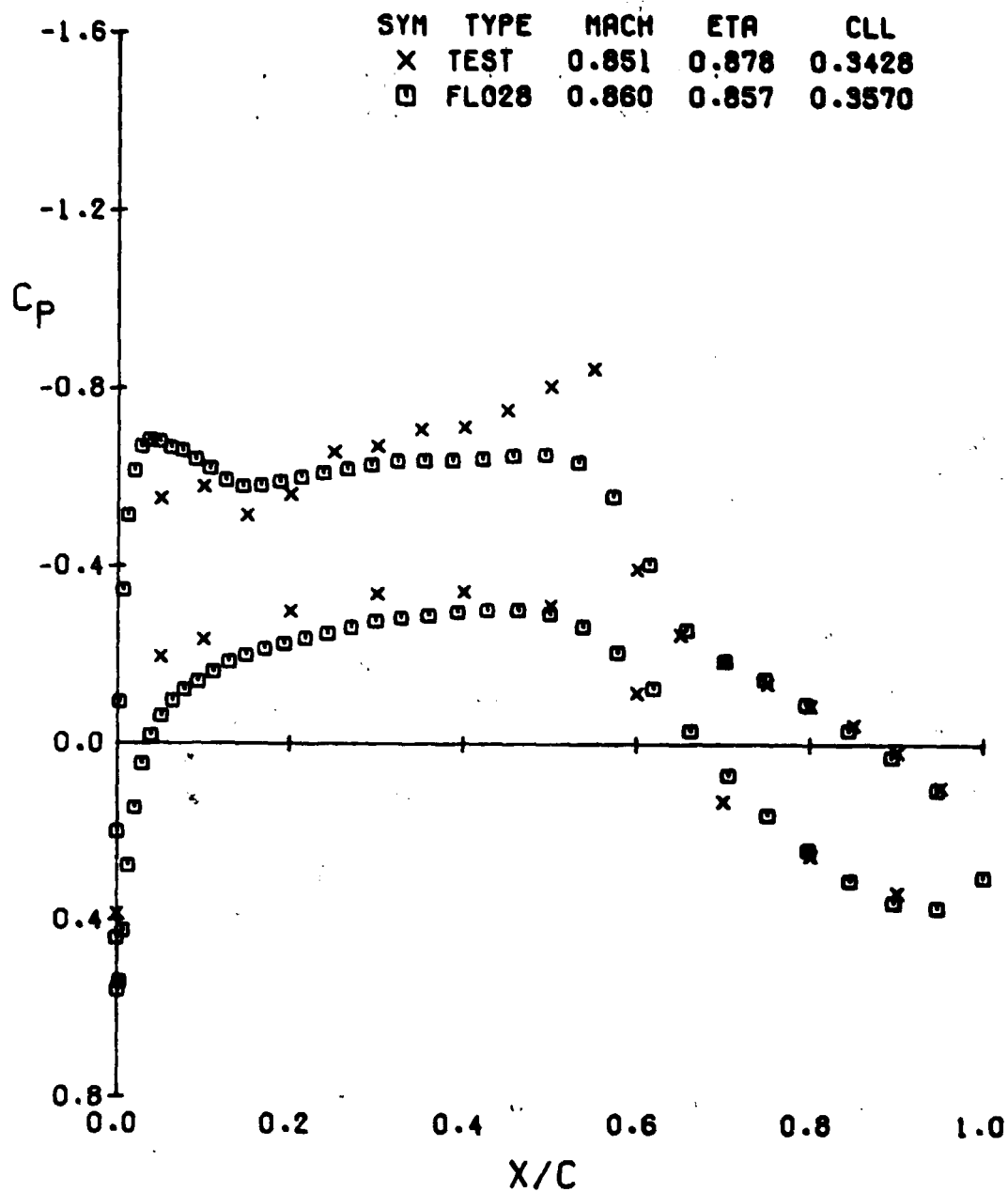
ISOLATED WING NO. 1



(C) $\text{ETA} = 0.634$

FIGURE 7.- CONTINUED

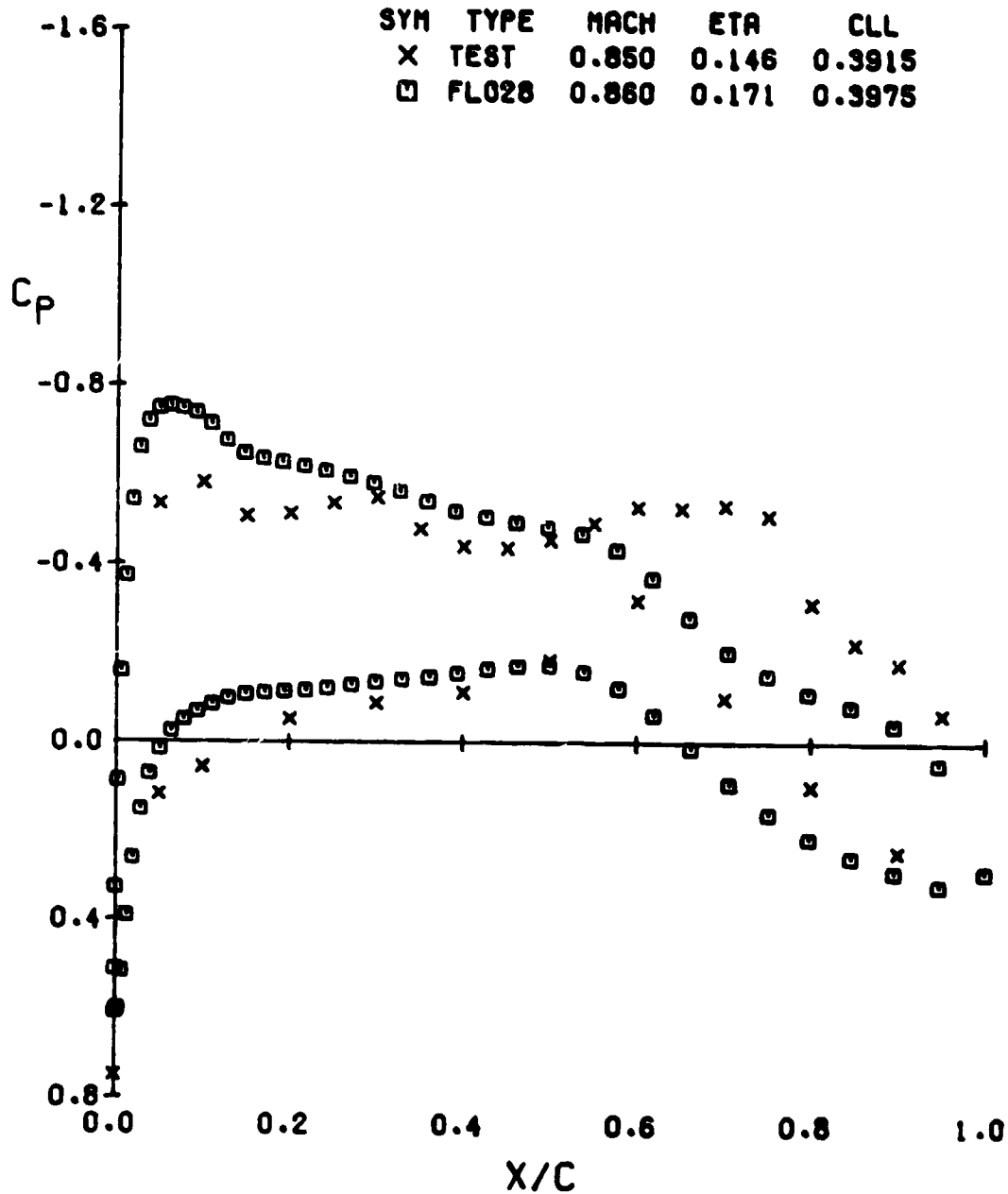
ISOLATED WING NO. 1



(D) $\text{ETA} = 0.878$

FIGURE 7.- CONCLUDED

ISOLATED WING NO. 1

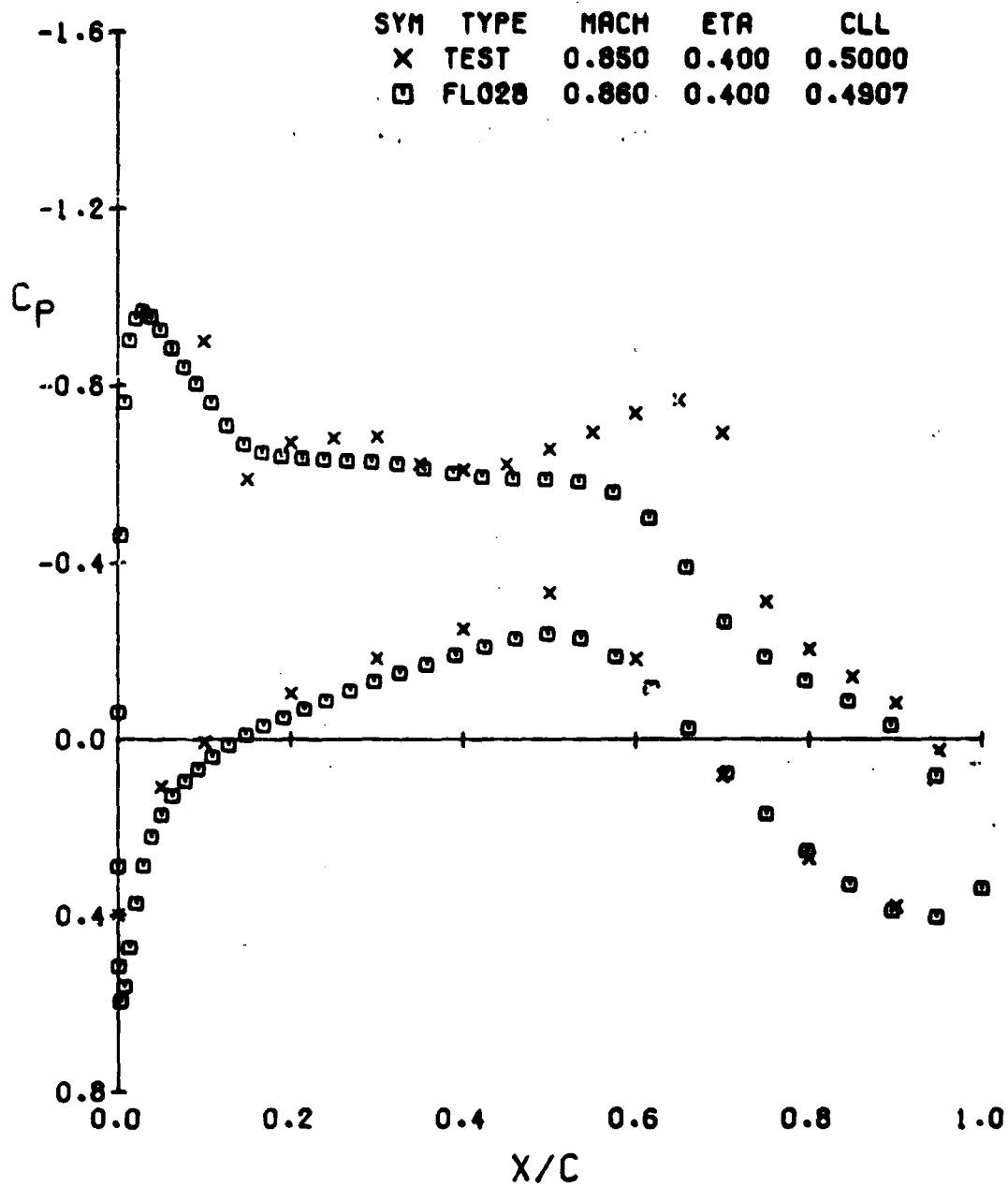


(A) $\eta = 0.146$

FIGURE 8.- COMPARISON OF FLO28 WING PRESSURES WITH EXPERIMENT

$\alpha = 6.52$ $CL(EXP) = .515$

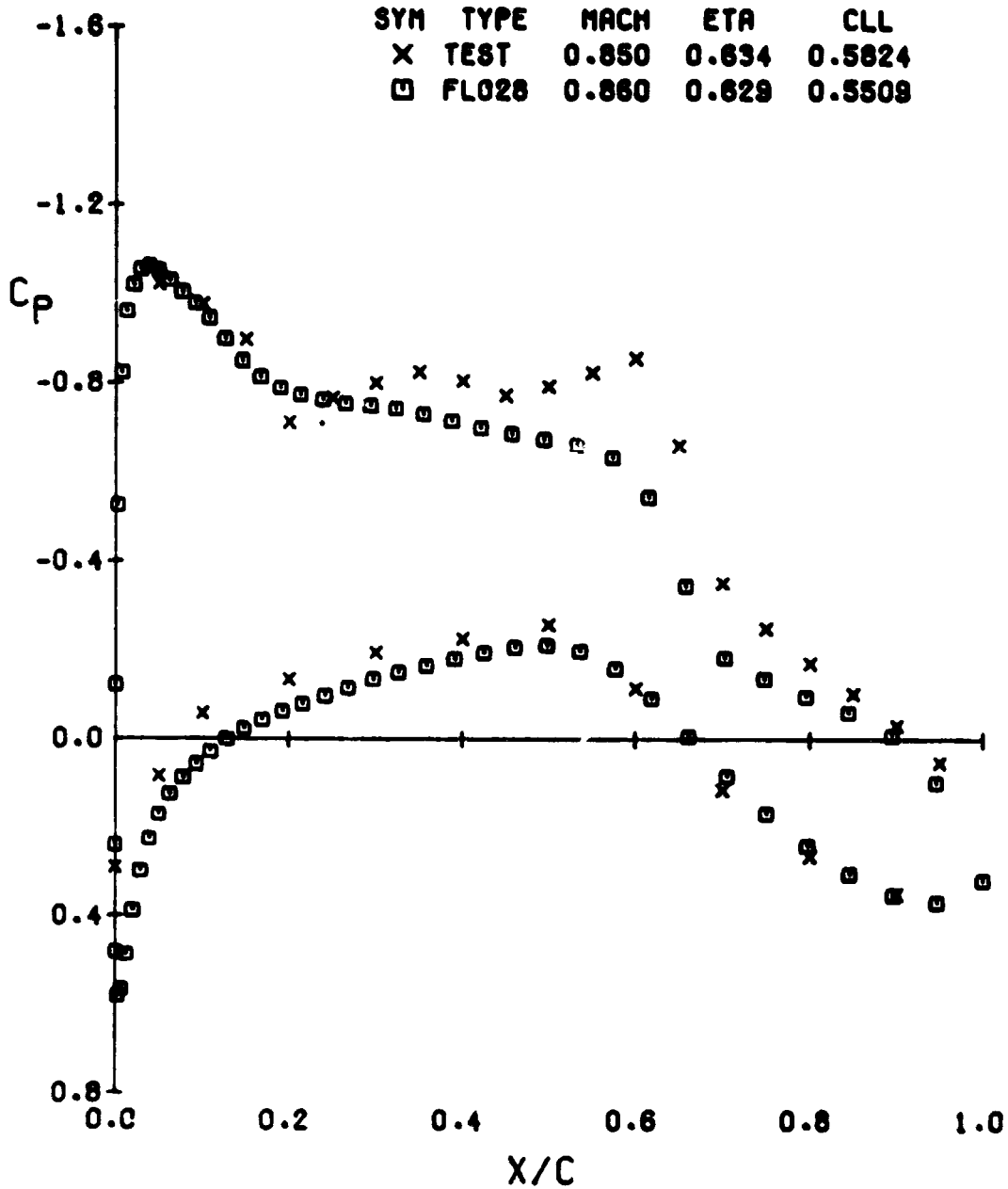
ISOLATED WING NO. 1



(B) $\text{ETA} = 0.400$

FIGURE 8.- CONTINUED

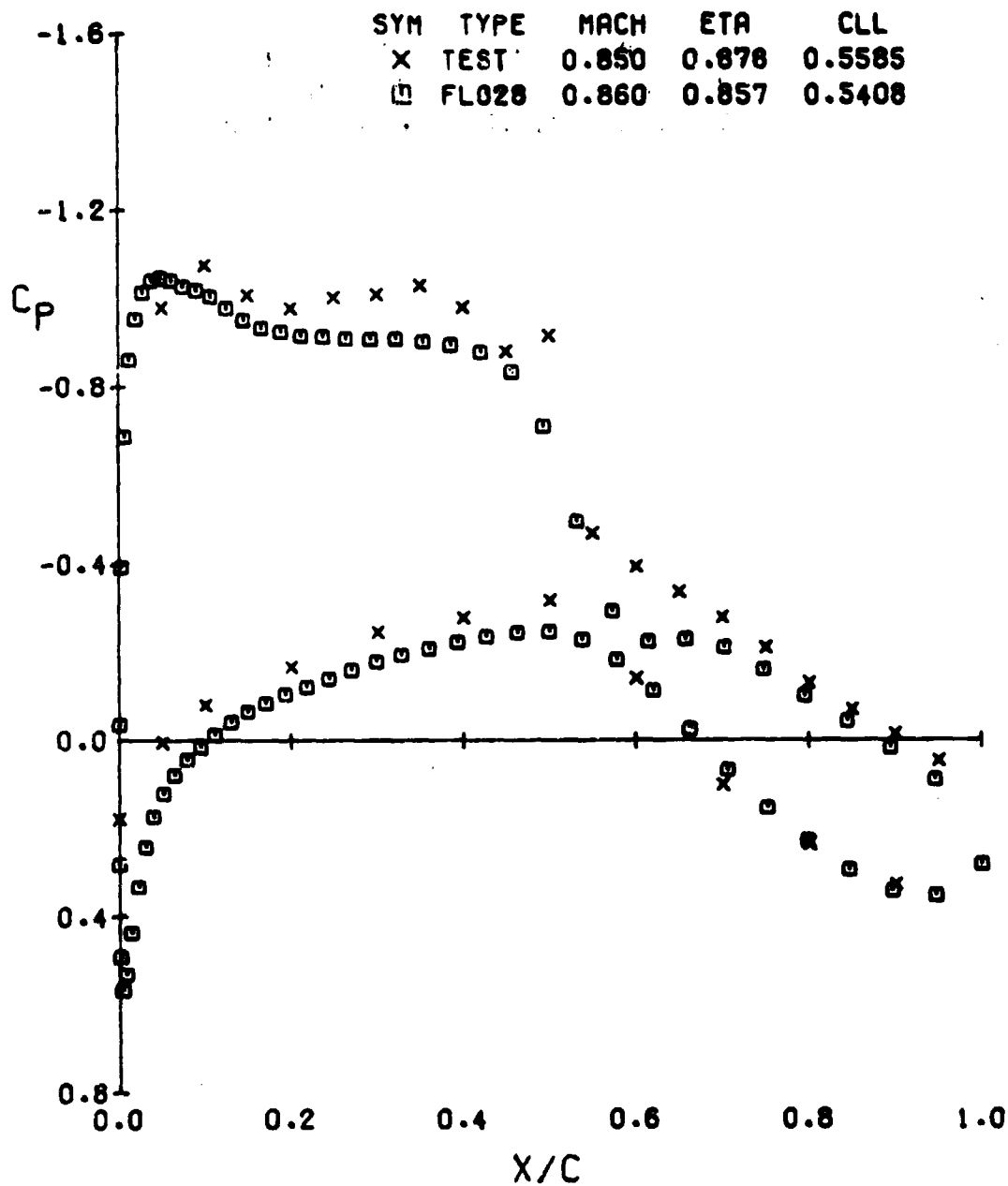
ISOLATED WING NO. 1



(C) $\text{ETA} = 0.634$

FIGURE 8.- CONTINUED

ISOLATED WING NO. 1



(□) $\eta = 0.878$

FIGURE 8.- CONCLUDED

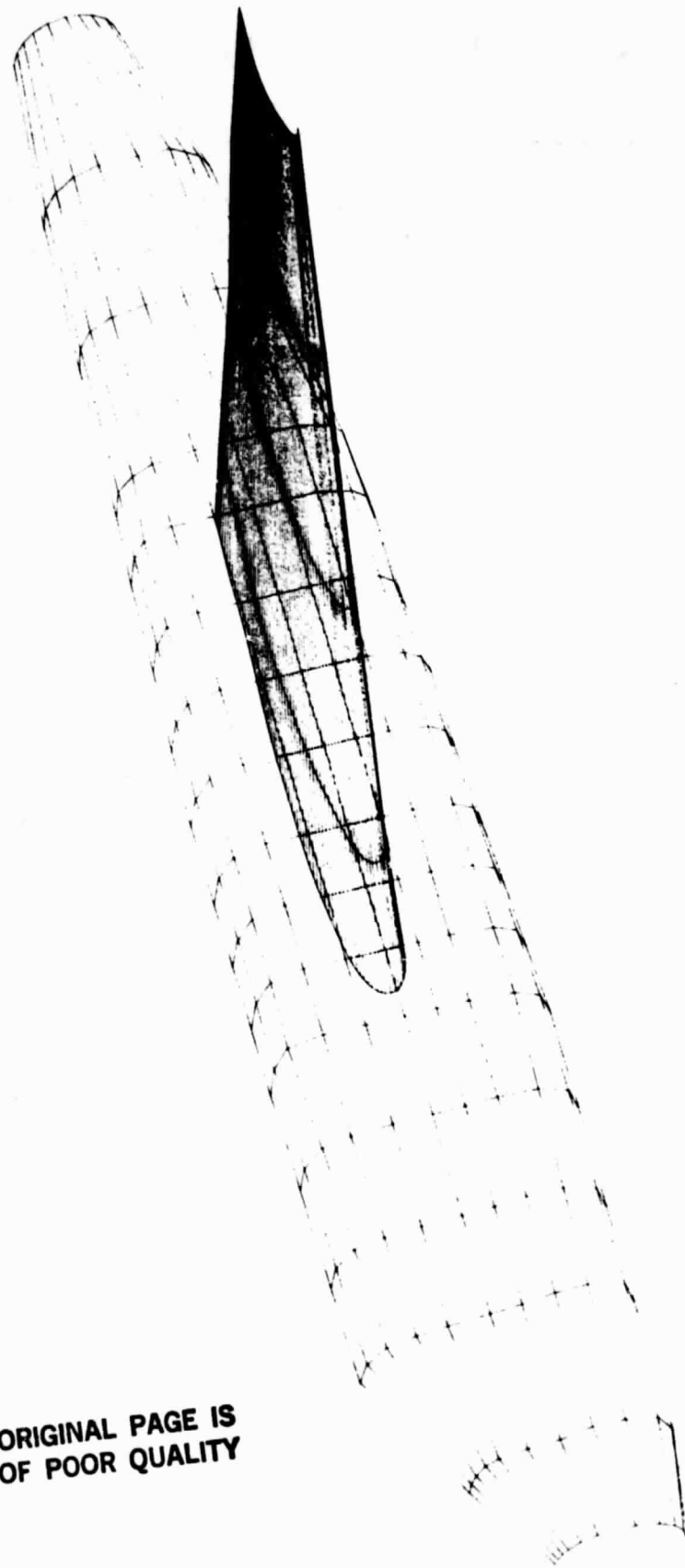
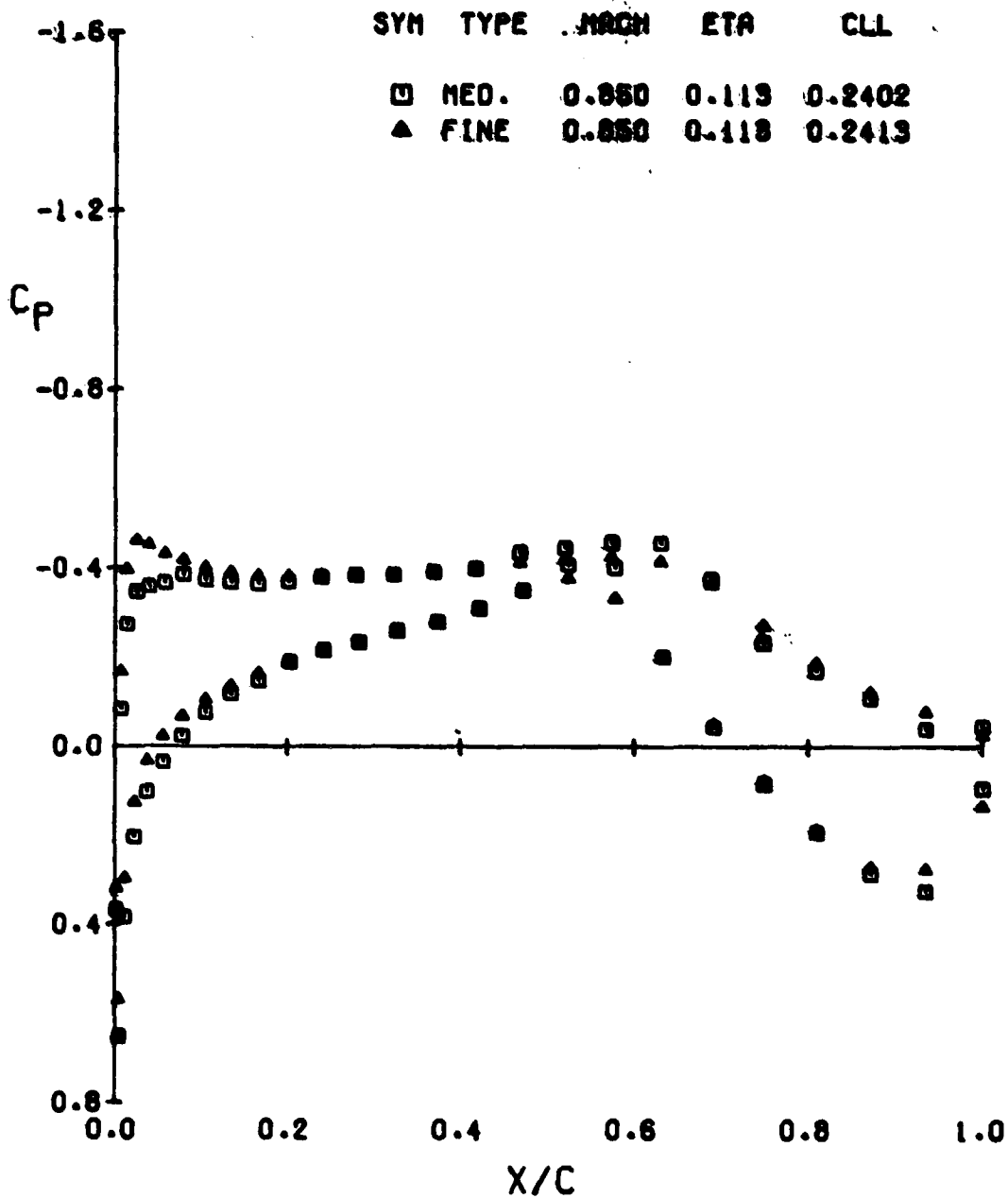


FIGURE 9.- A-7 TRANSONIC WING-BODY GEOMETRY

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OF POOR QUALITY

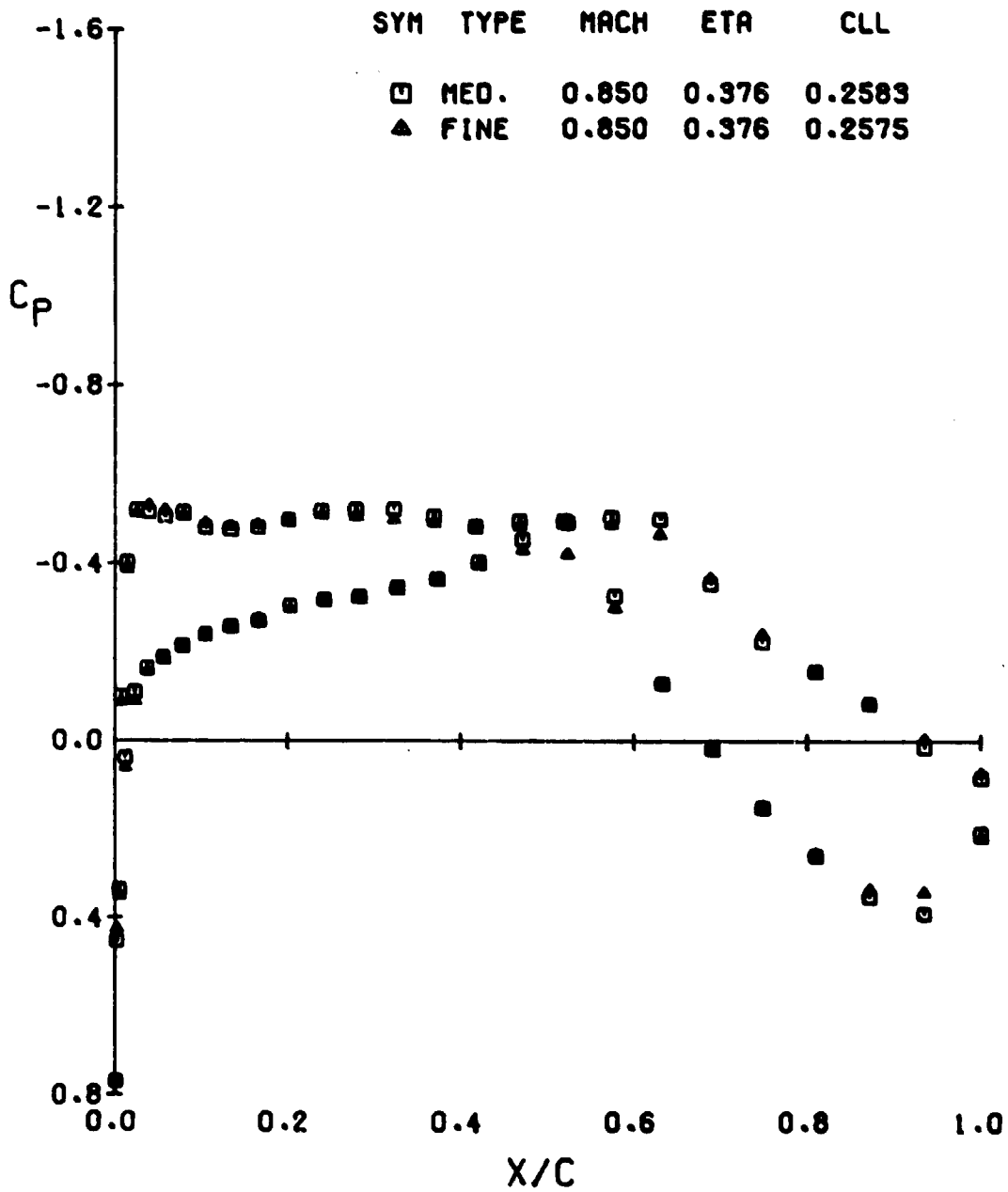
BODY/WING NO.2 T/C=.09



(A) ETA = 0.113

FIGURE 10.- COMPARISON OF MEDIUM AND FINE GRID SOLUTIONS
 FLO30 MED = 80X12X16 FINE = 160X24X32

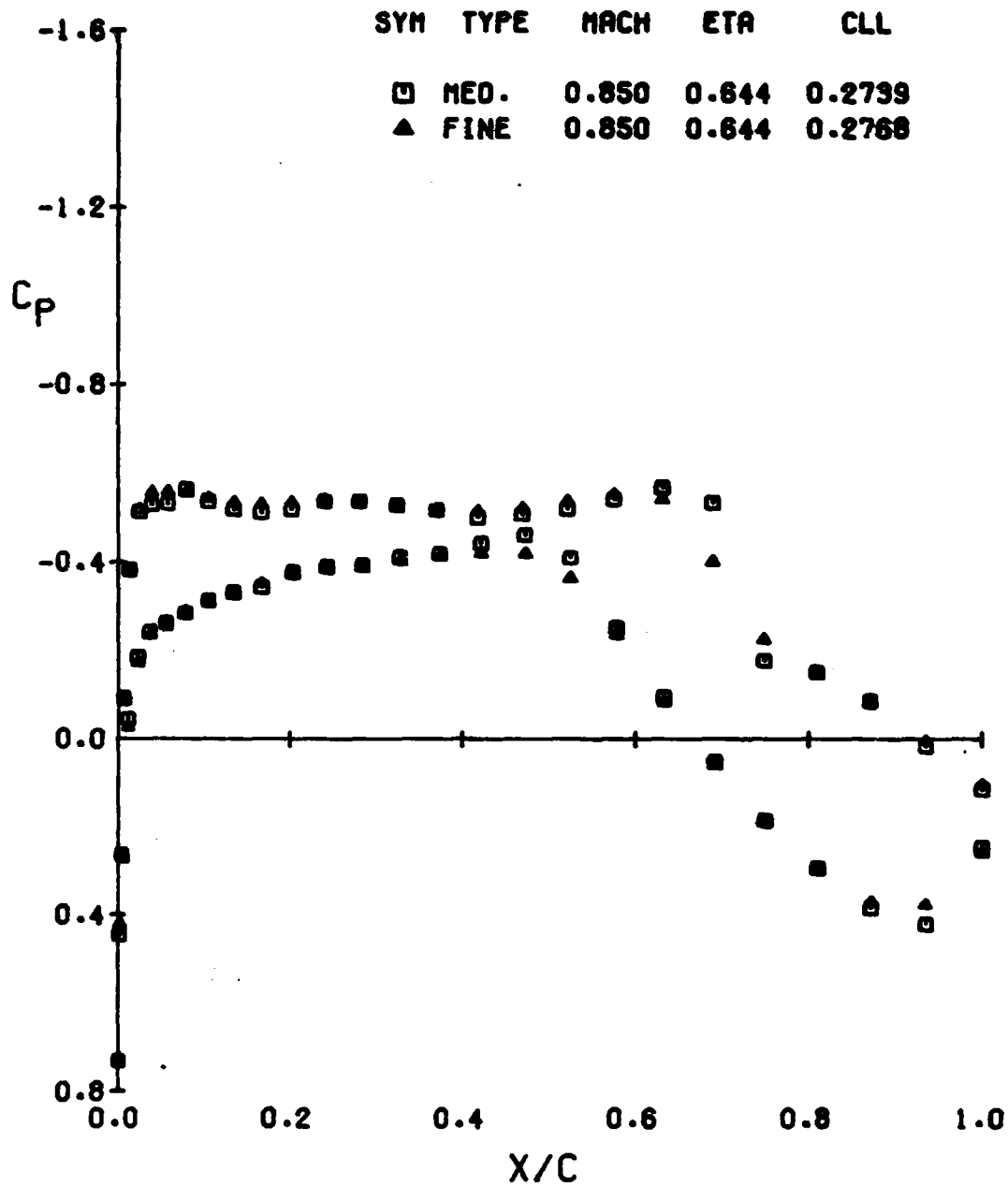
BODY/WING NO.2 T/C=.09



(B) $\eta = 0.376$

FIGURE 10.- CONTINUED

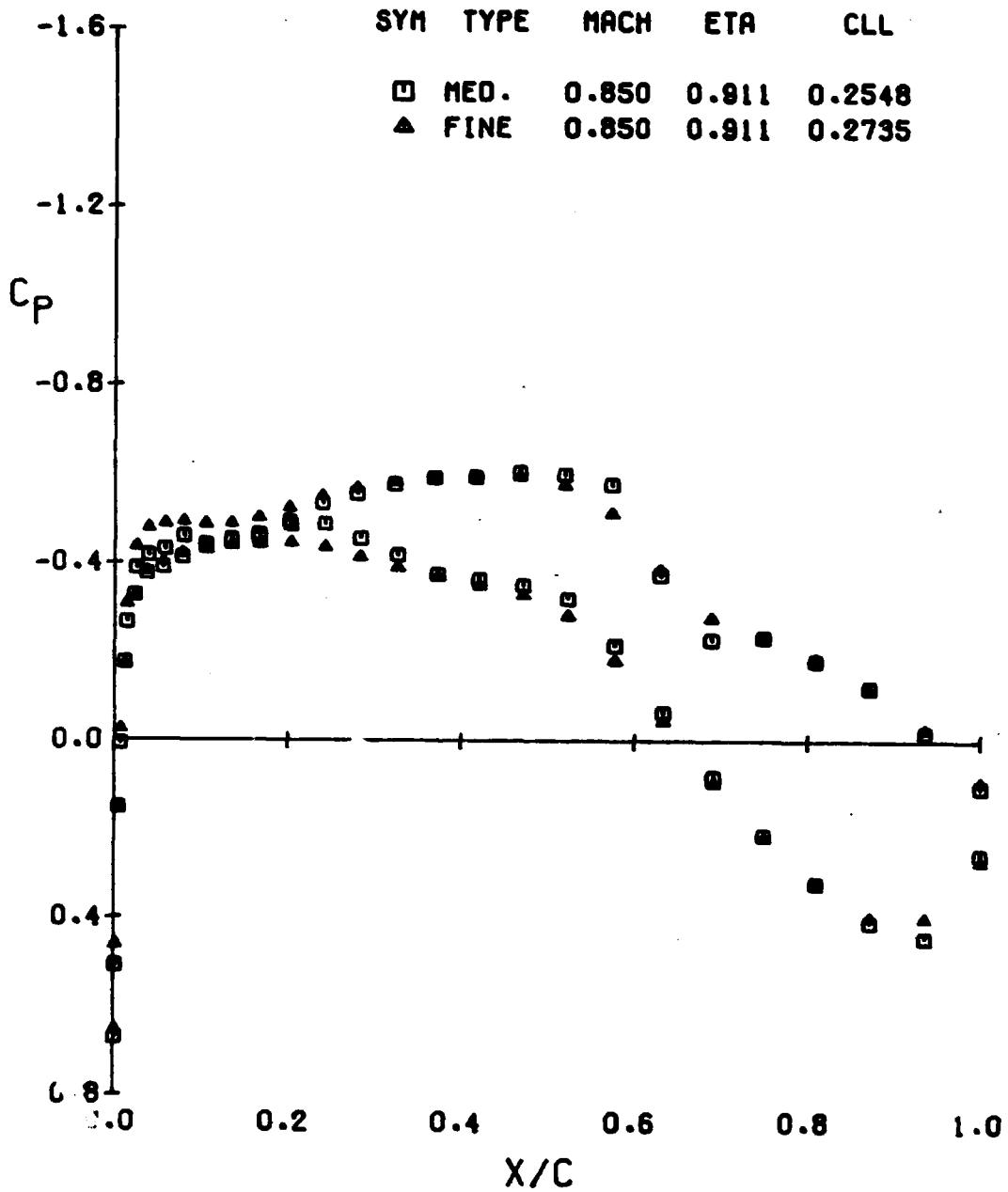
BODY/WING NO.2 T/C=.09



(C) $\eta = 0.644$

FIGURE 10.- CONTINUED

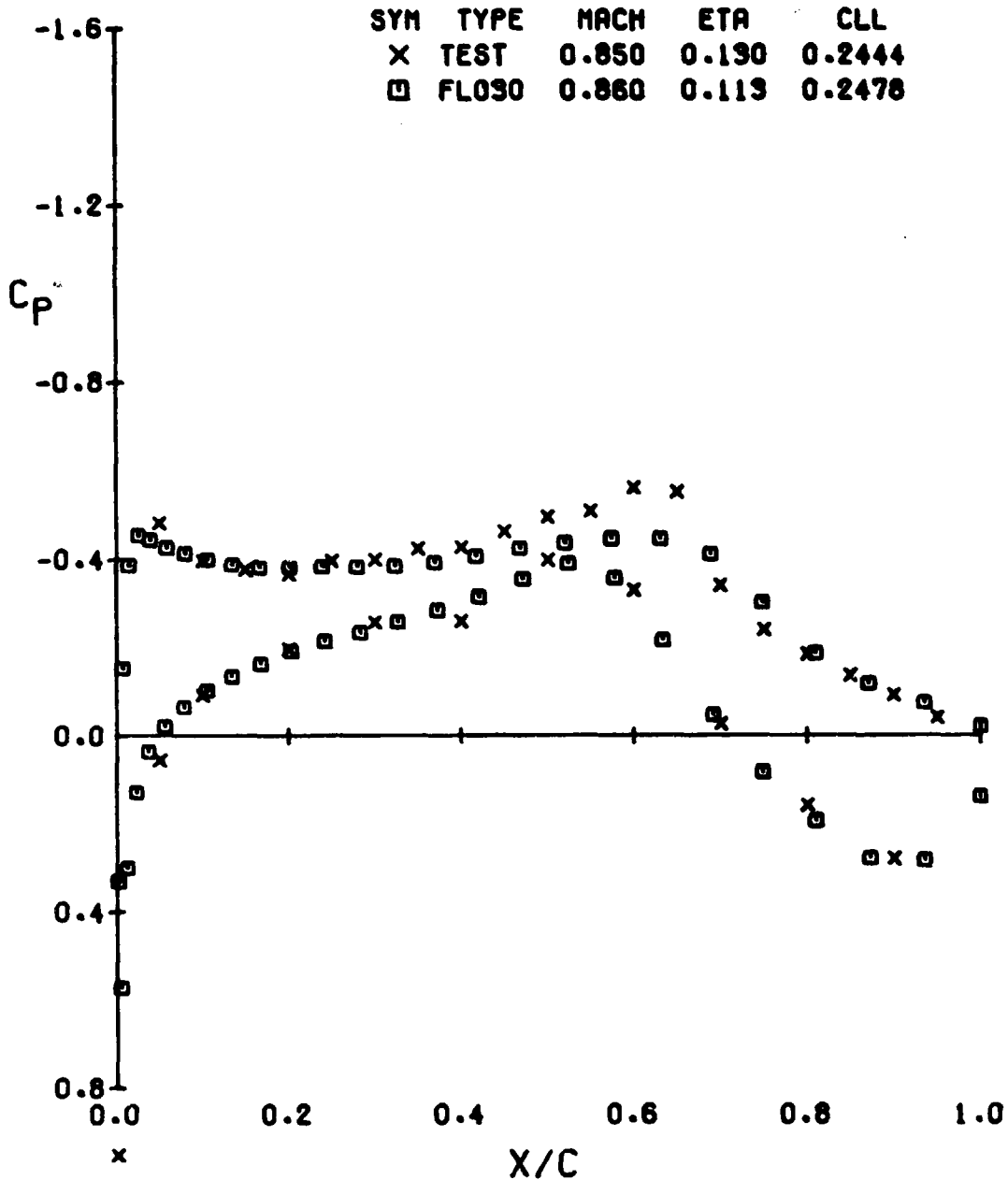
BODY/WING NO.2 T/C=.09



(D) ETA = 0.911

FIGURE 10.- CONCLUDED

BODY/WING NO.2 T/C=.09

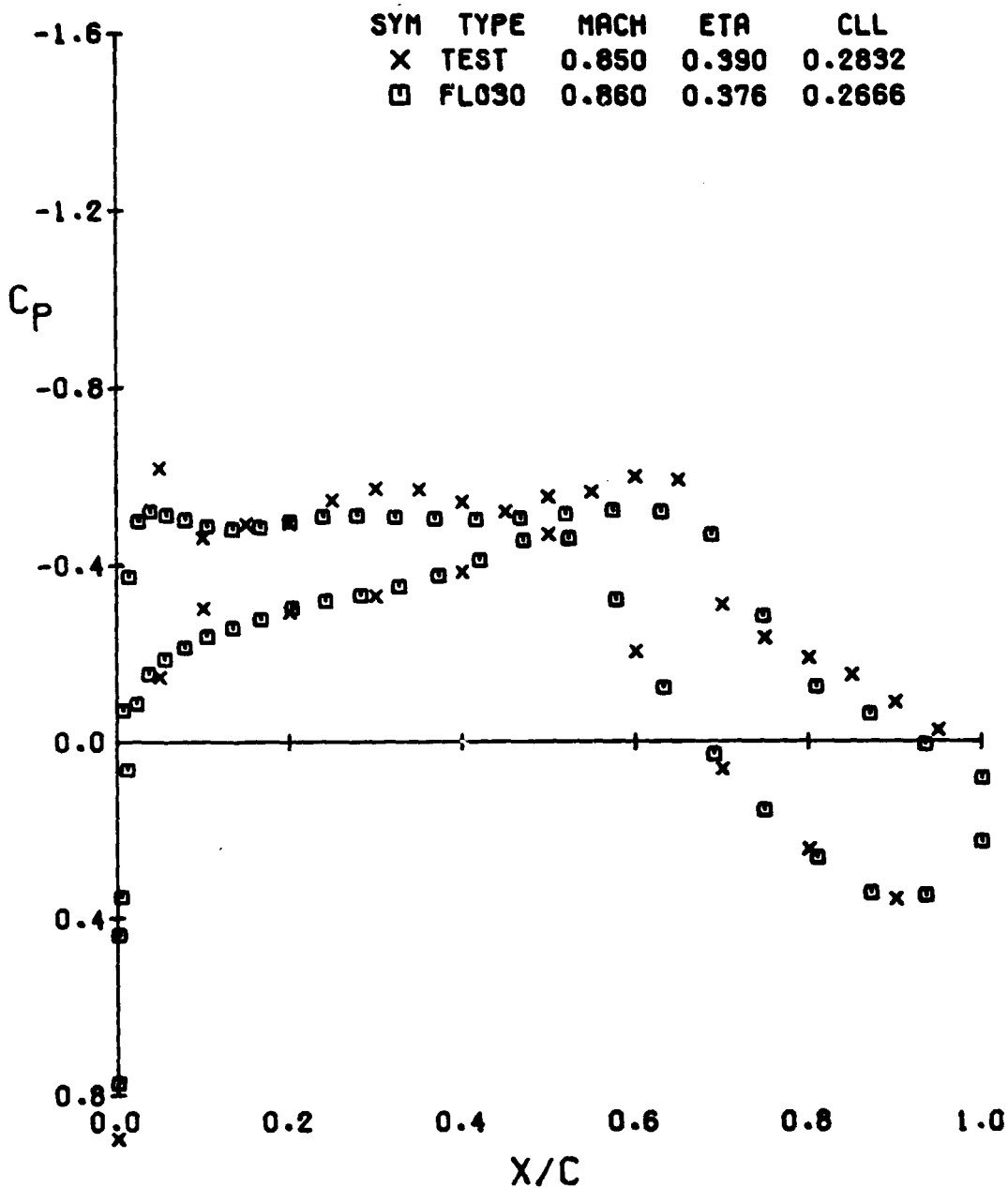


(A) ETA = 0.190

FIGURE 11.- COMPARISON OF FLO30 WING/BODY PRESSURES WITH EXPERIMENT

ALPHA = 4.48 CL(EXP) = .282

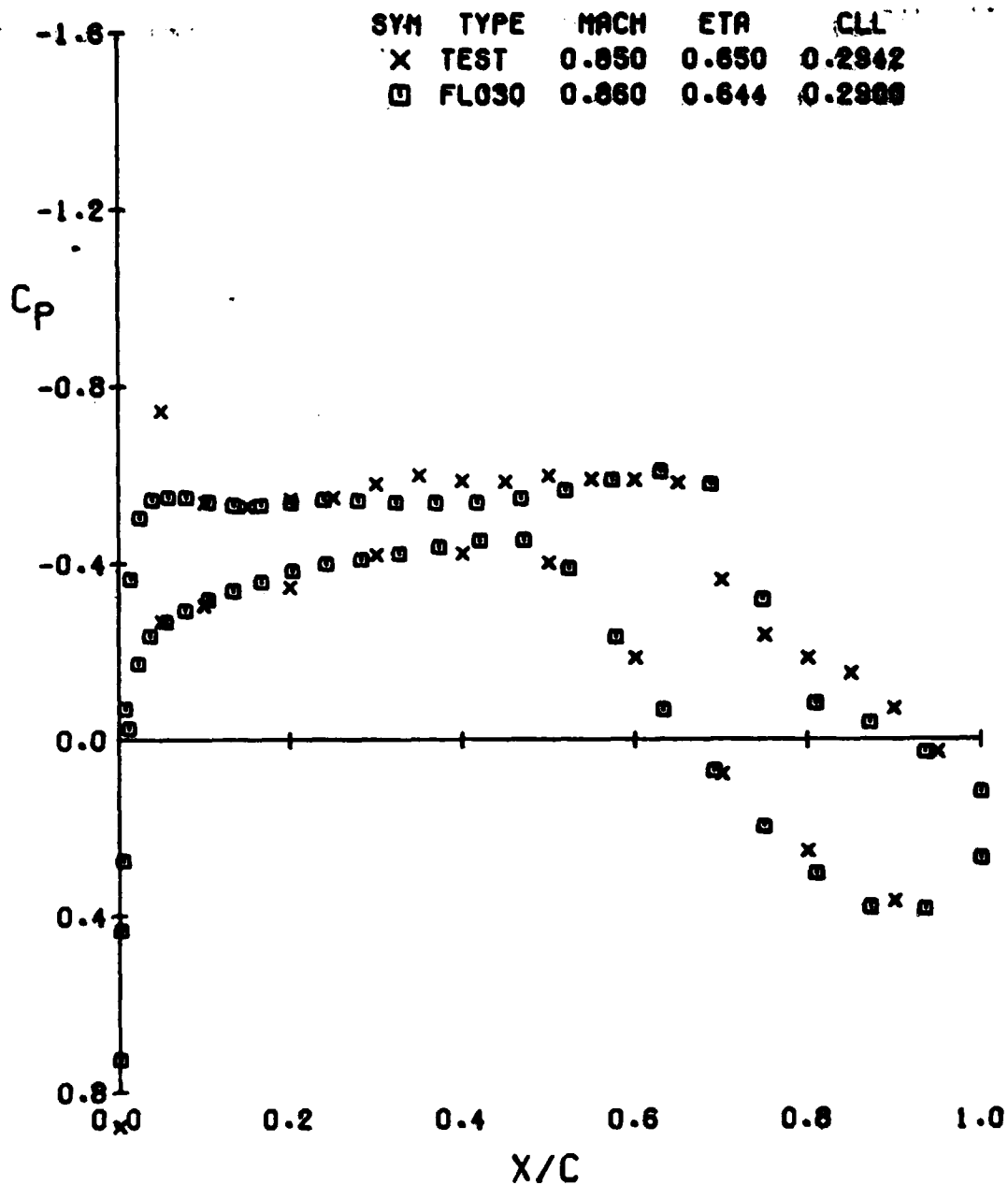
BODY/WING NO.2 T/C=.09



(B) ETA = 0.390

FIGURE 11.- CONTINUED

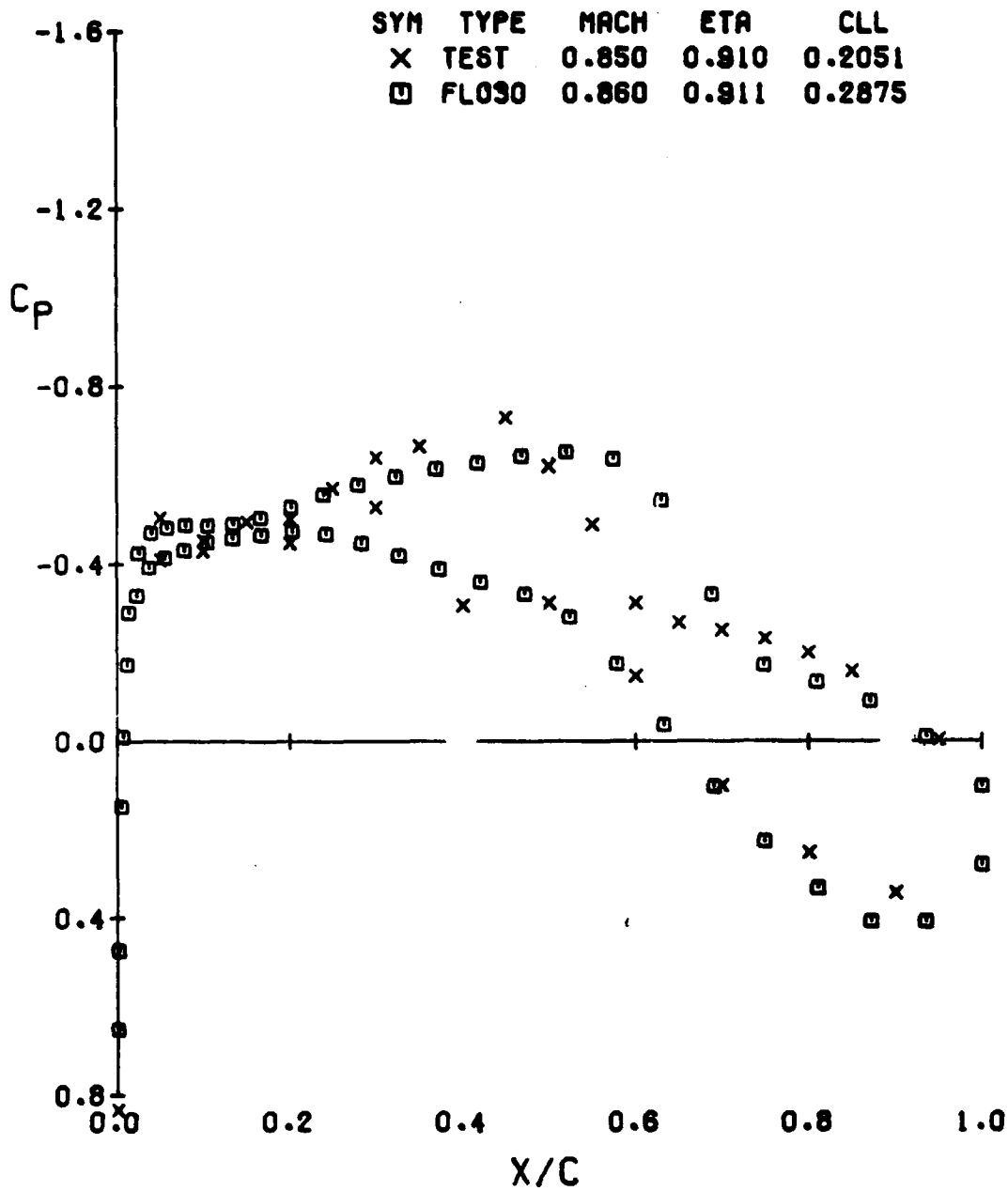
BODY/WING NO.2 T/C=.09



(C) ETA = 0.650

FIGURE 11.- CONTINUED

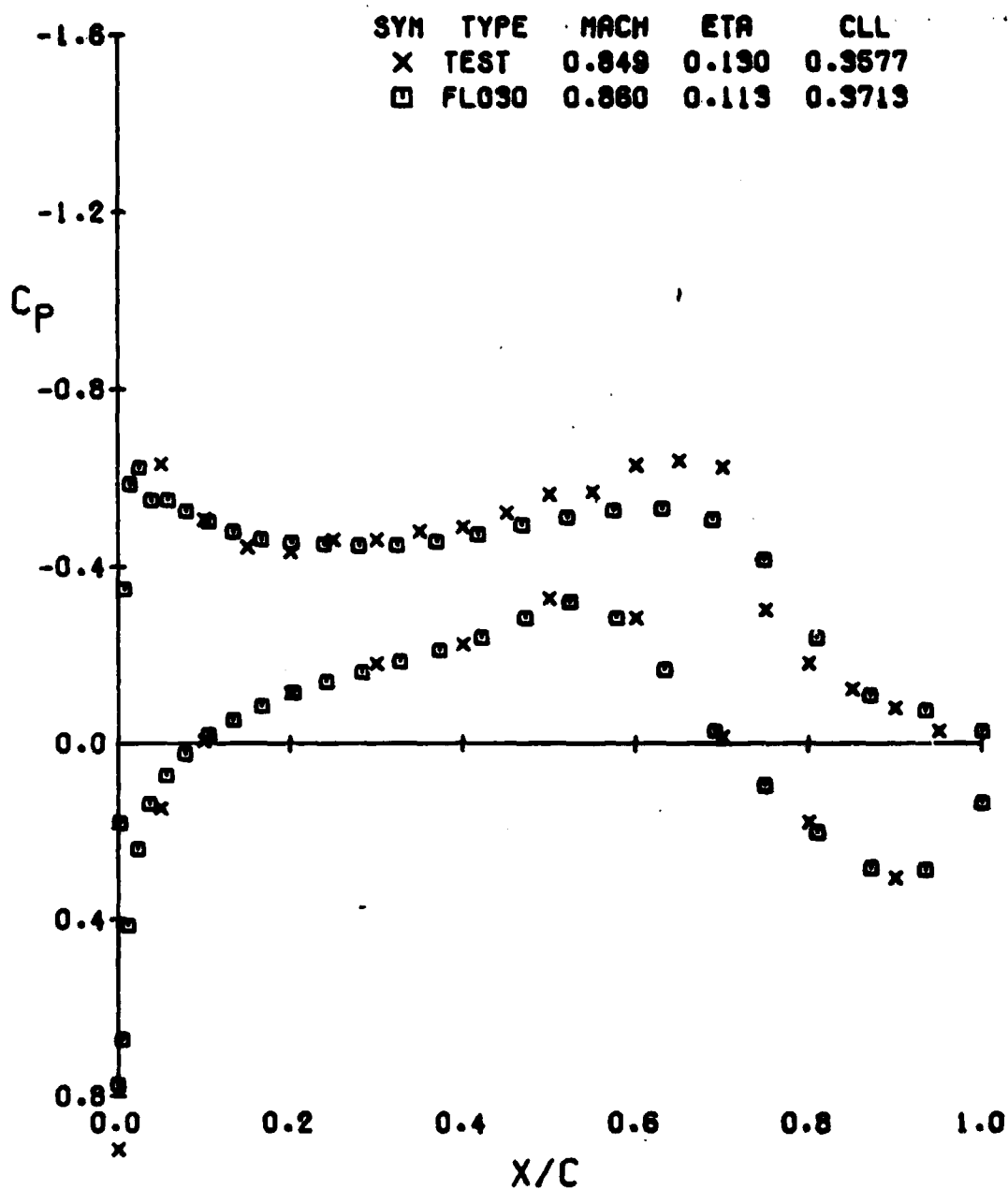
BODY/WING NO.2 T/C=.09



(D) ETA = 0.910

FIGURE 11.- CONCLUDED

BODY/WING NO.2 T/C=.09

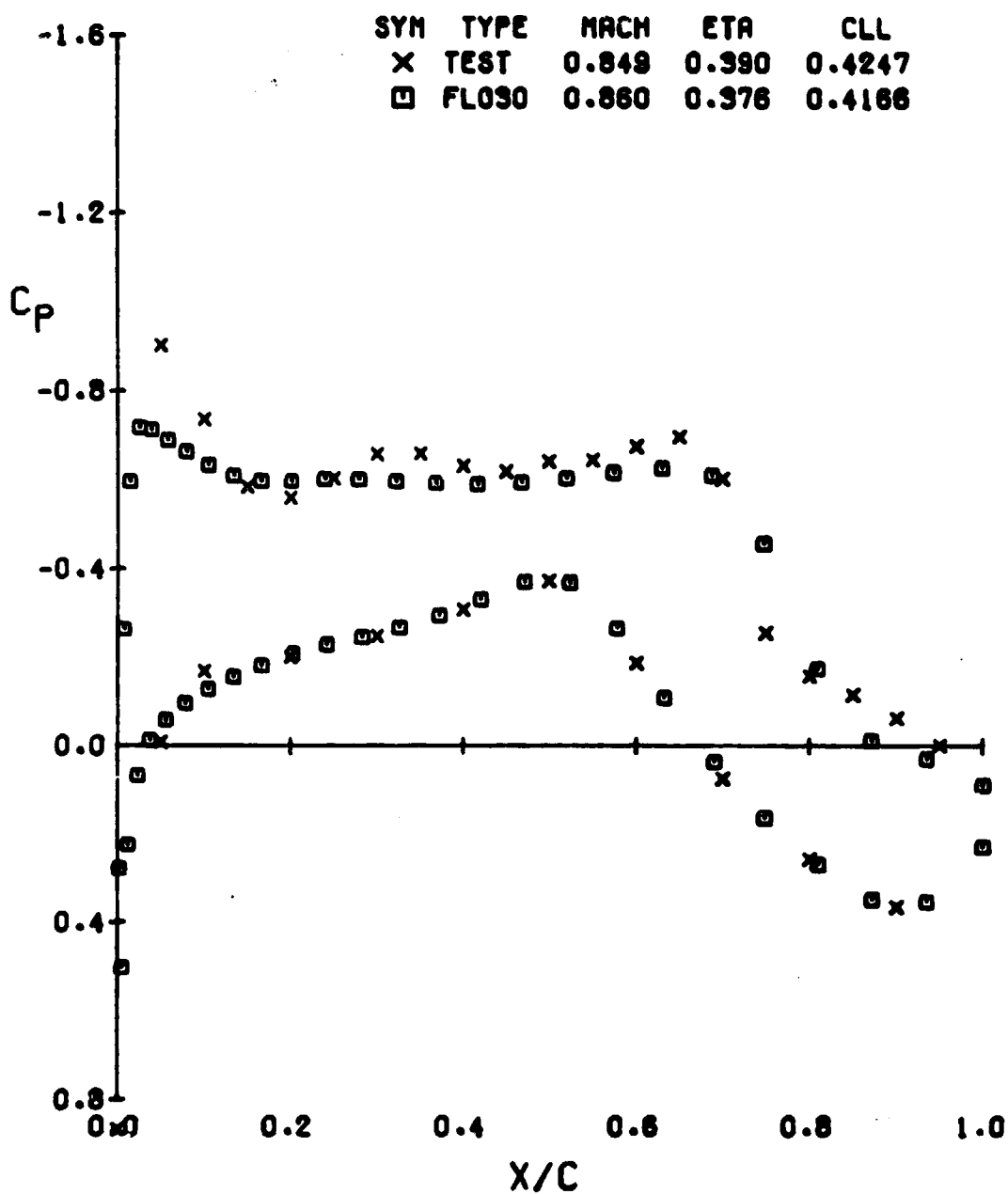


(A) ETA = 0.130

FIGURE 12.- COMPARISON OF FLO30 WING/BODY PRESSURES WITH EXPERIMENT

ALPHA = 5.62 CL(EXP) = .416

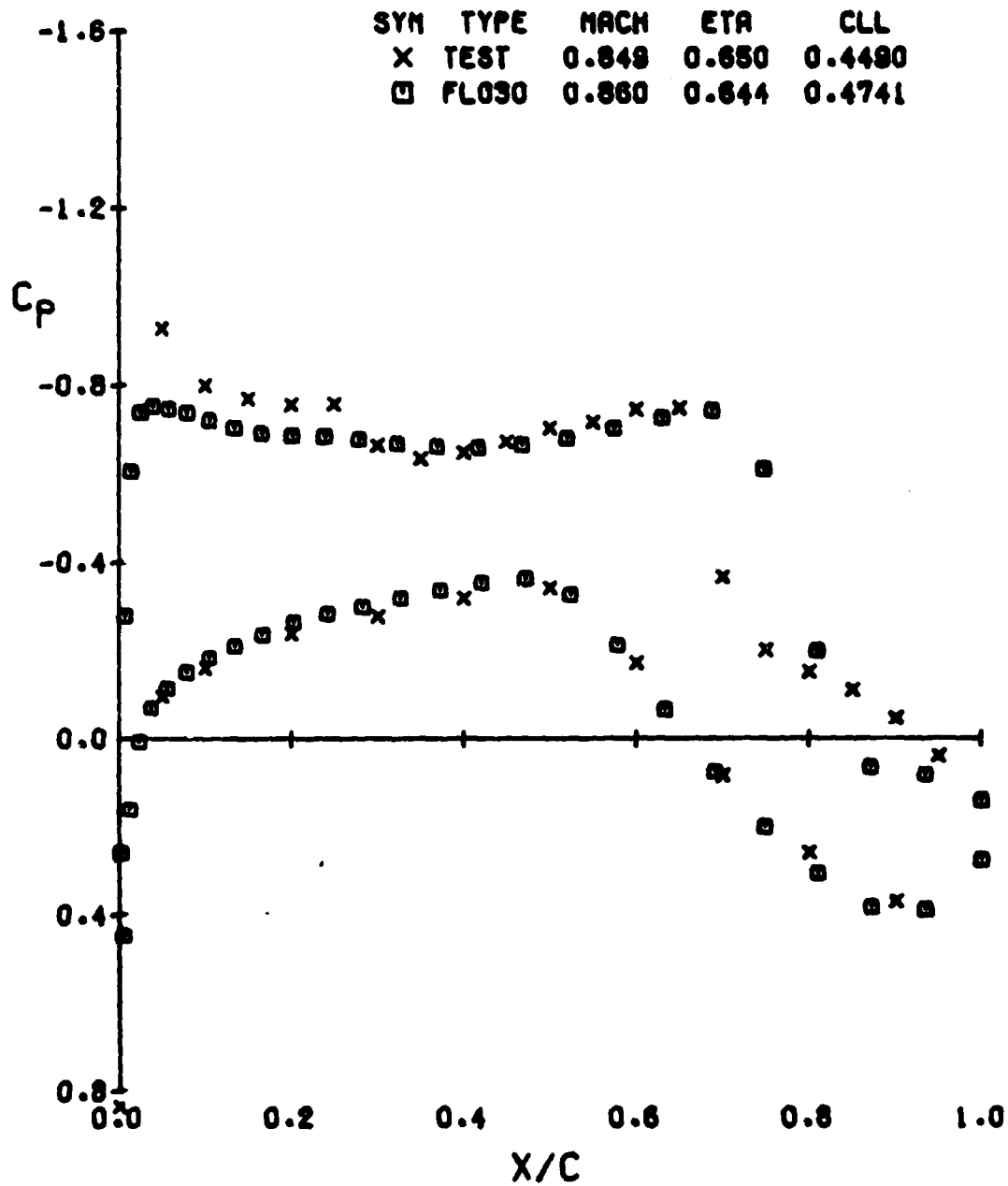
BODY/WING NO.2 T/C=.09



(B) ETA = 0.390

FIGURE 12.- CONTINUED

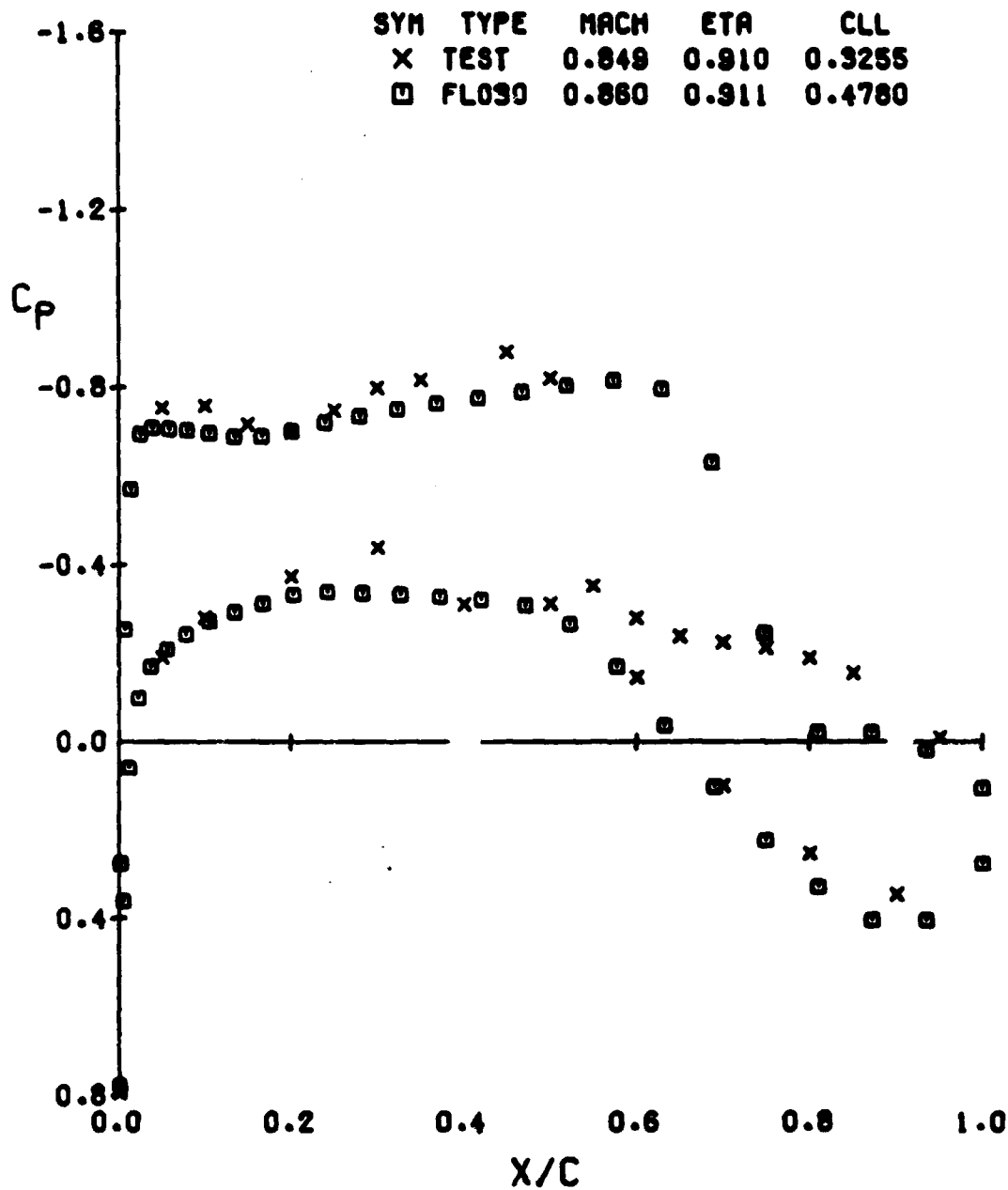
BODY/WING NO.2 T/C=.09



(C) ETA = 0.650

FIGURE 12.- CONTINUED

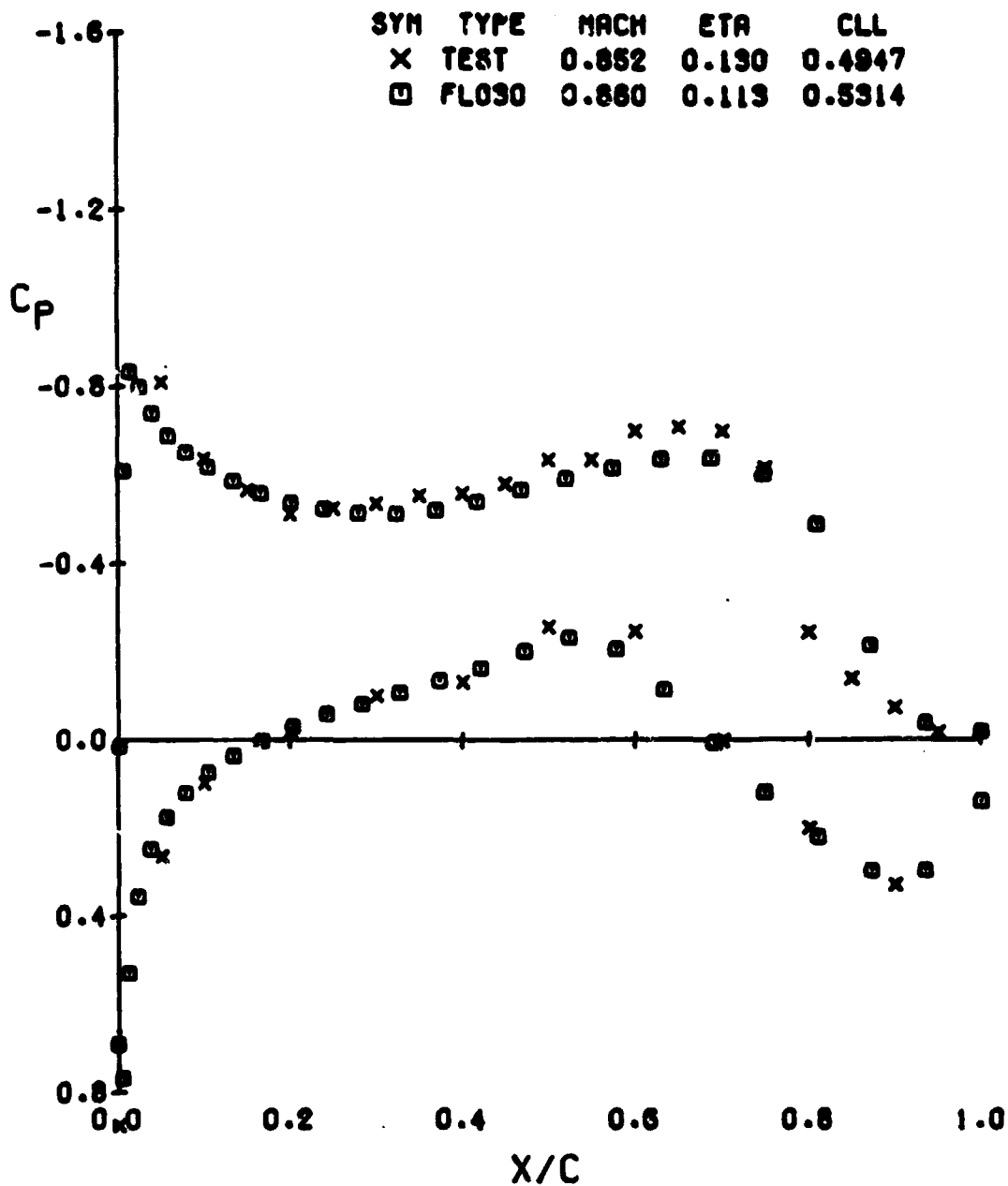
BODY/WING NO.2 T/C=.09



(D) ETA = 0.910

FIGURE 12.- CONCLUDED

BODY/WING NO.2 T/C=.09

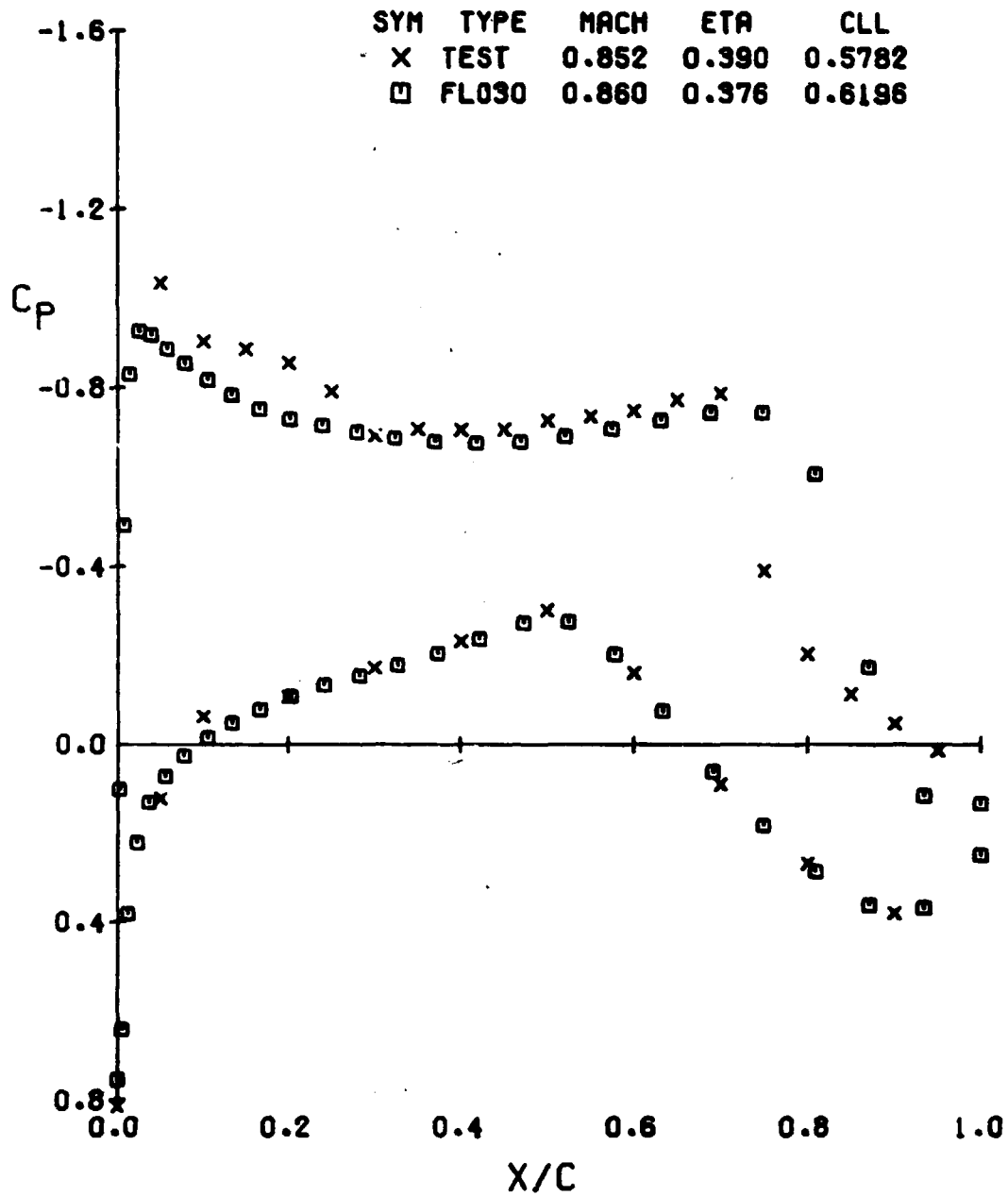


(A) ETA = 0.130

FIGURE 13.- COMPARISON OF FLO30 WING/BODY PRESSURES WITH EXPERIMENT

ALPHA = 6.8 CL(EXP) = .57

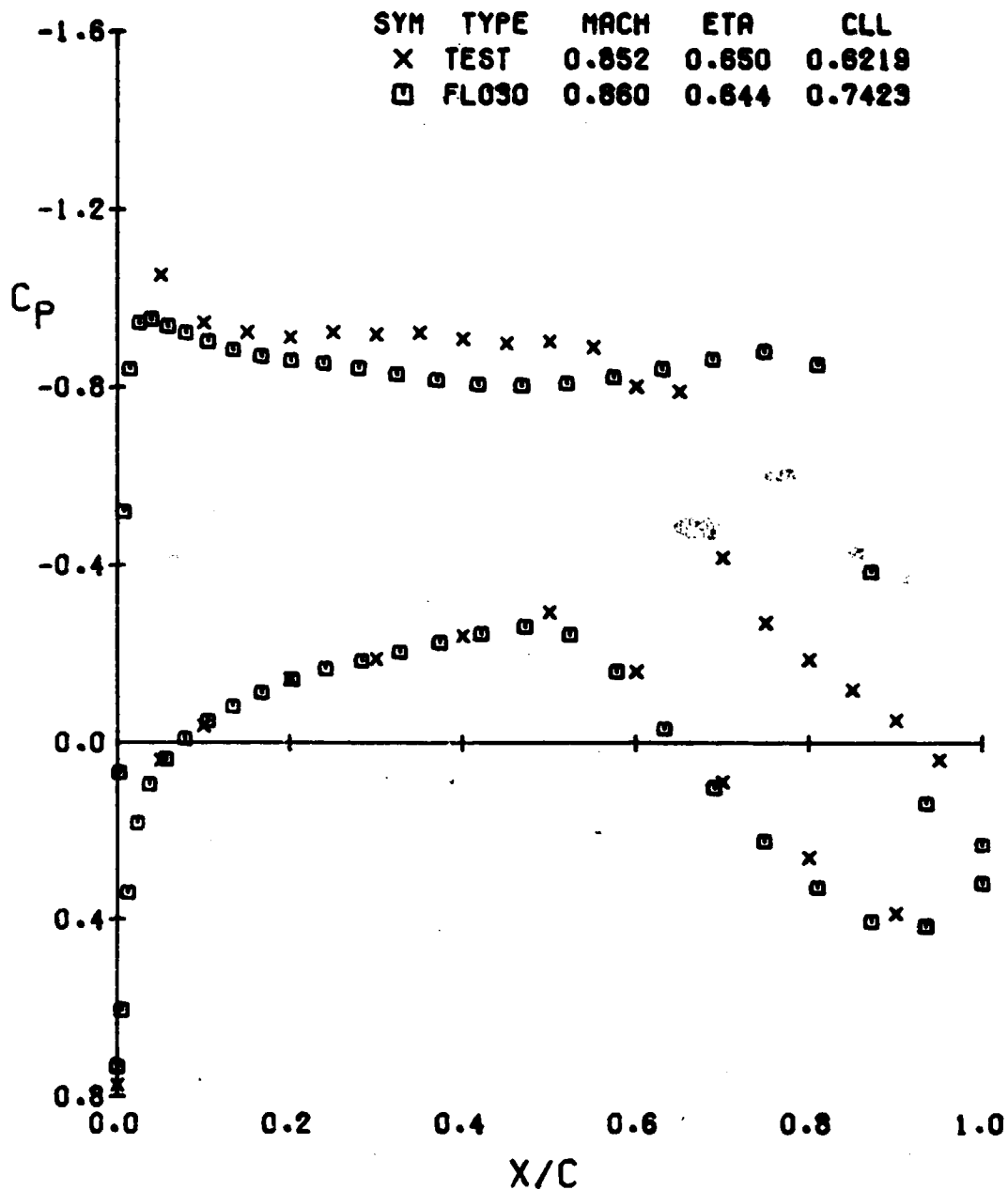
BODY/WING NO.2 T/C=.09



(B) ETA = 0.390

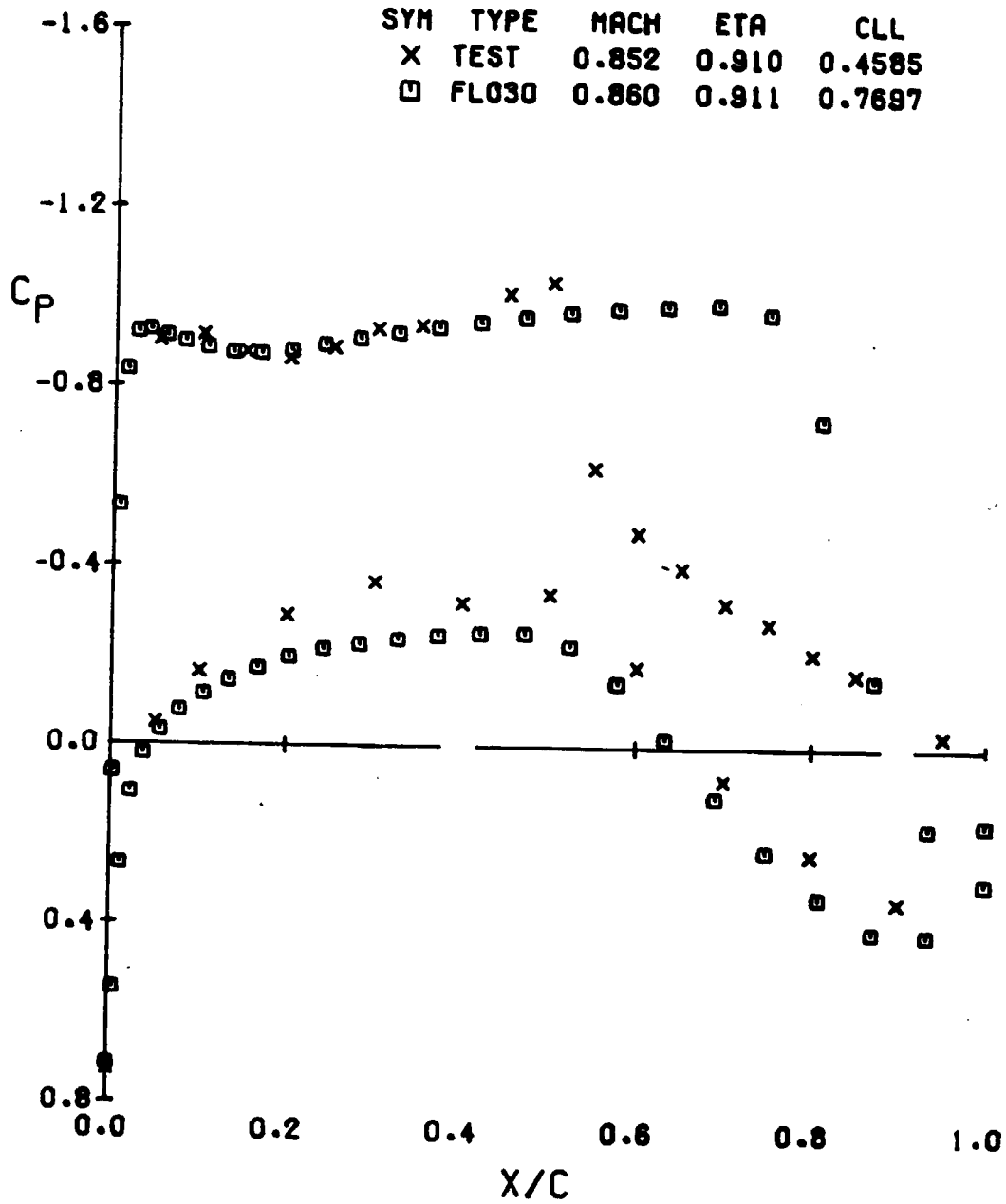
FIGURE 19.- CONTINUED

BODY/WING NO.2 T/C=.09



(C) ETA = 0.650
FIGURE 13.- CONTINUED

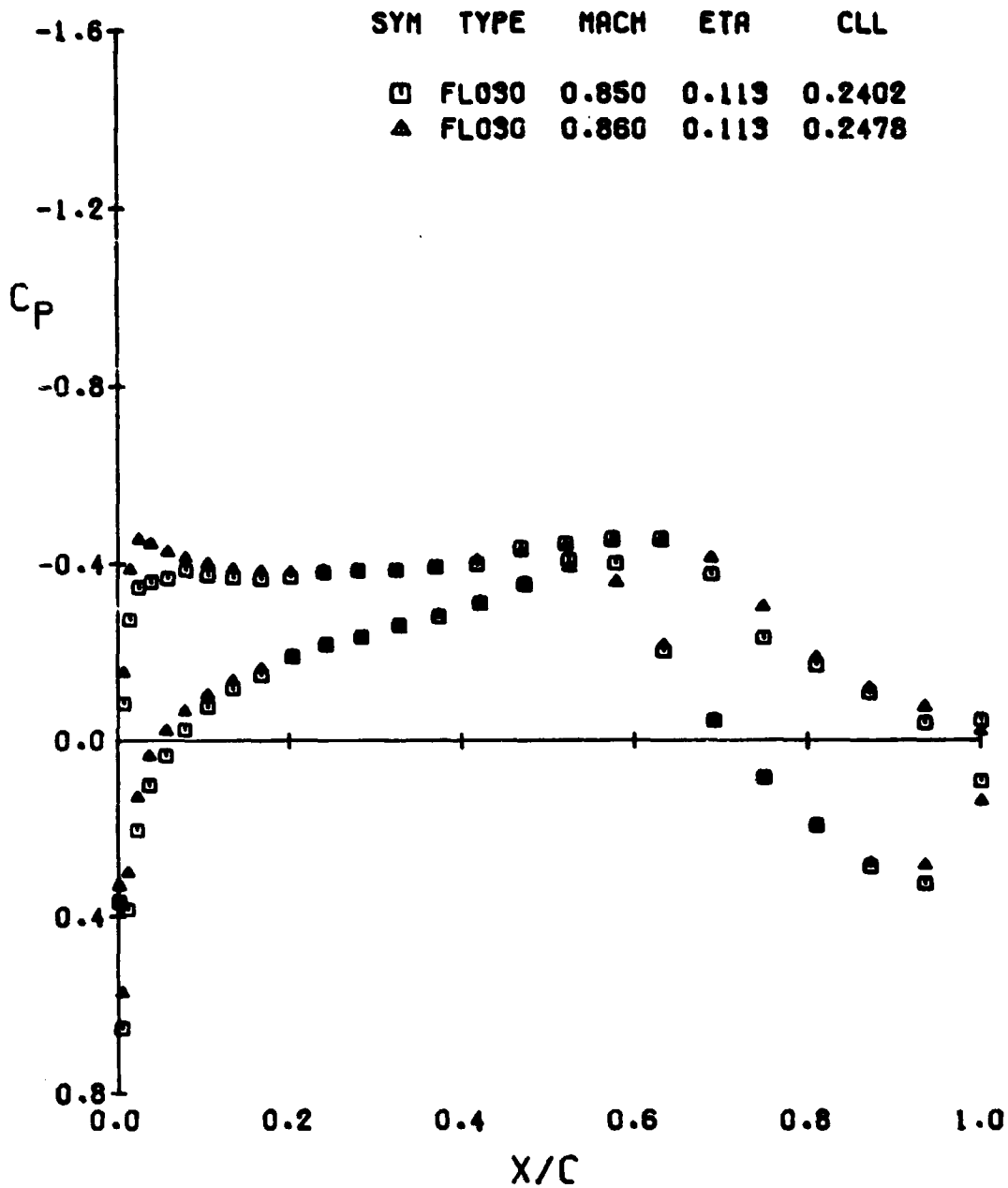
BODY/WING NO.2 T/C=.09



(O) ETA = 0.910

FIGURE 13.- CONCLUDED

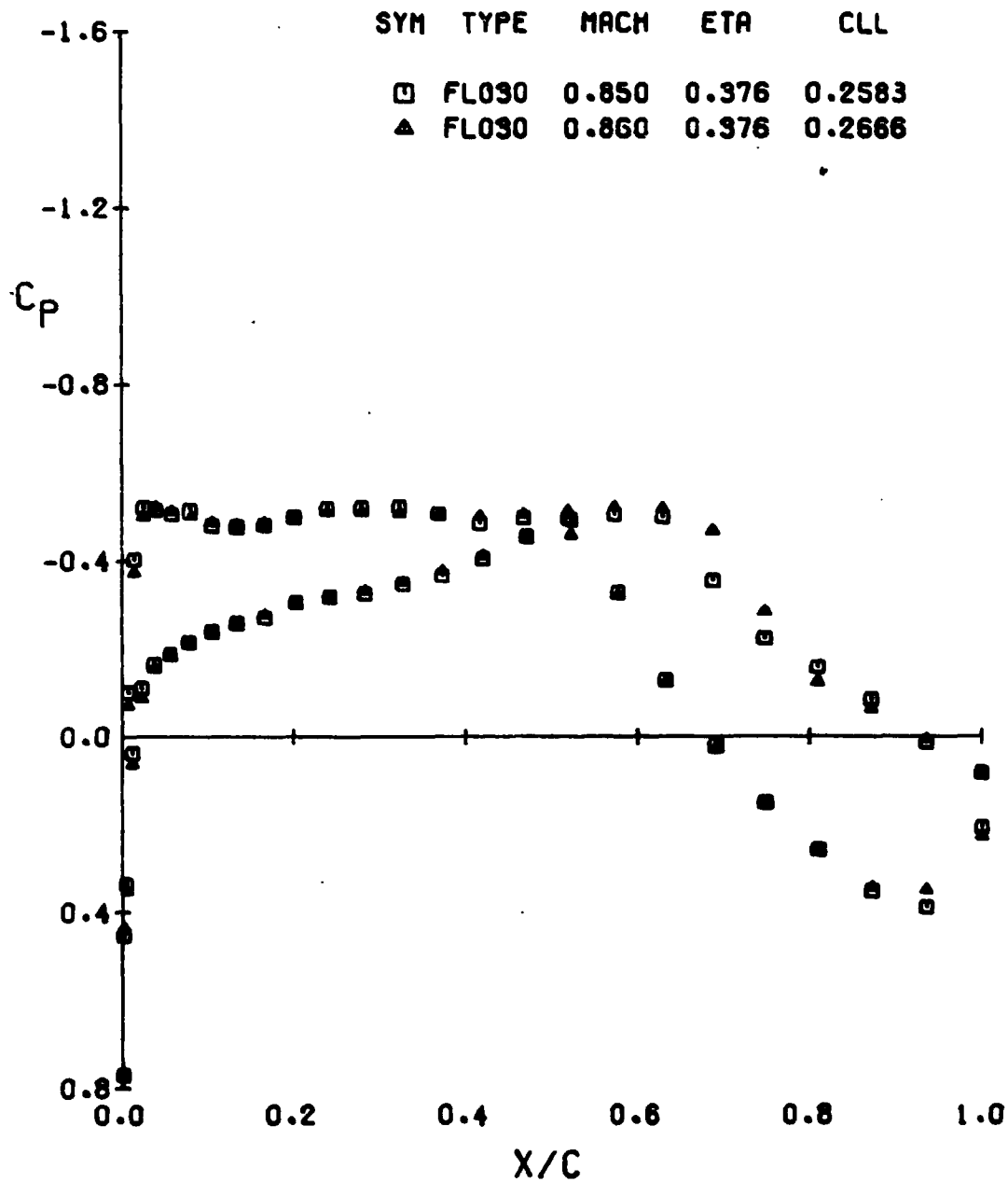
BODY/WING NO.2 T/C=.09



(A) $\eta = 0.113$

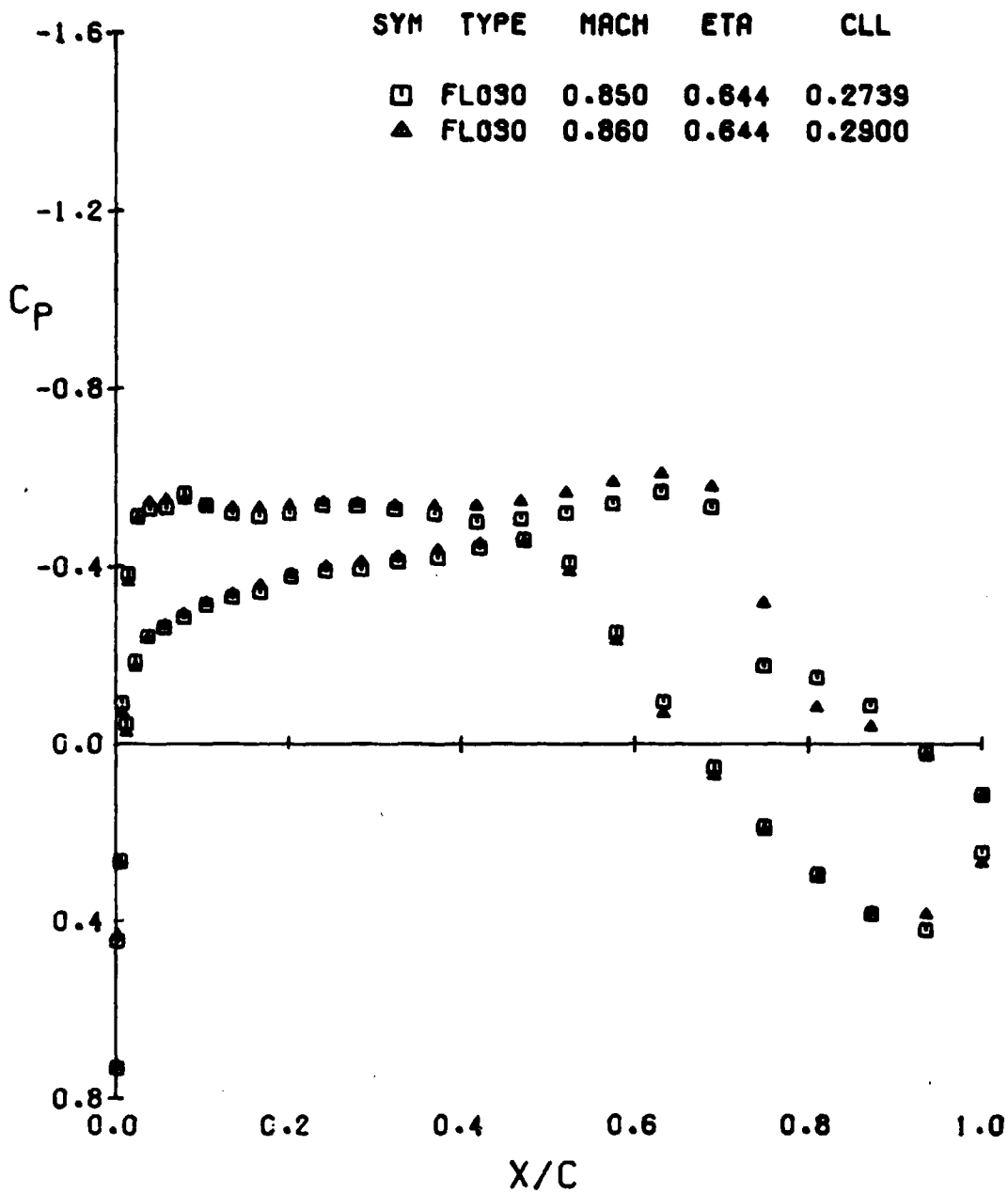
FIGURE 14.- EFFECT OF 0.01 MACH NUMBER CHANGE ON PREDICTED PRESSURES
CAUGHEY FLO90 TRANSONIC ANALYSIS

BODY/WING NO.2 T/C=.09



(B) $\eta = 0.376$
FIGURE 14.- CONTINUED

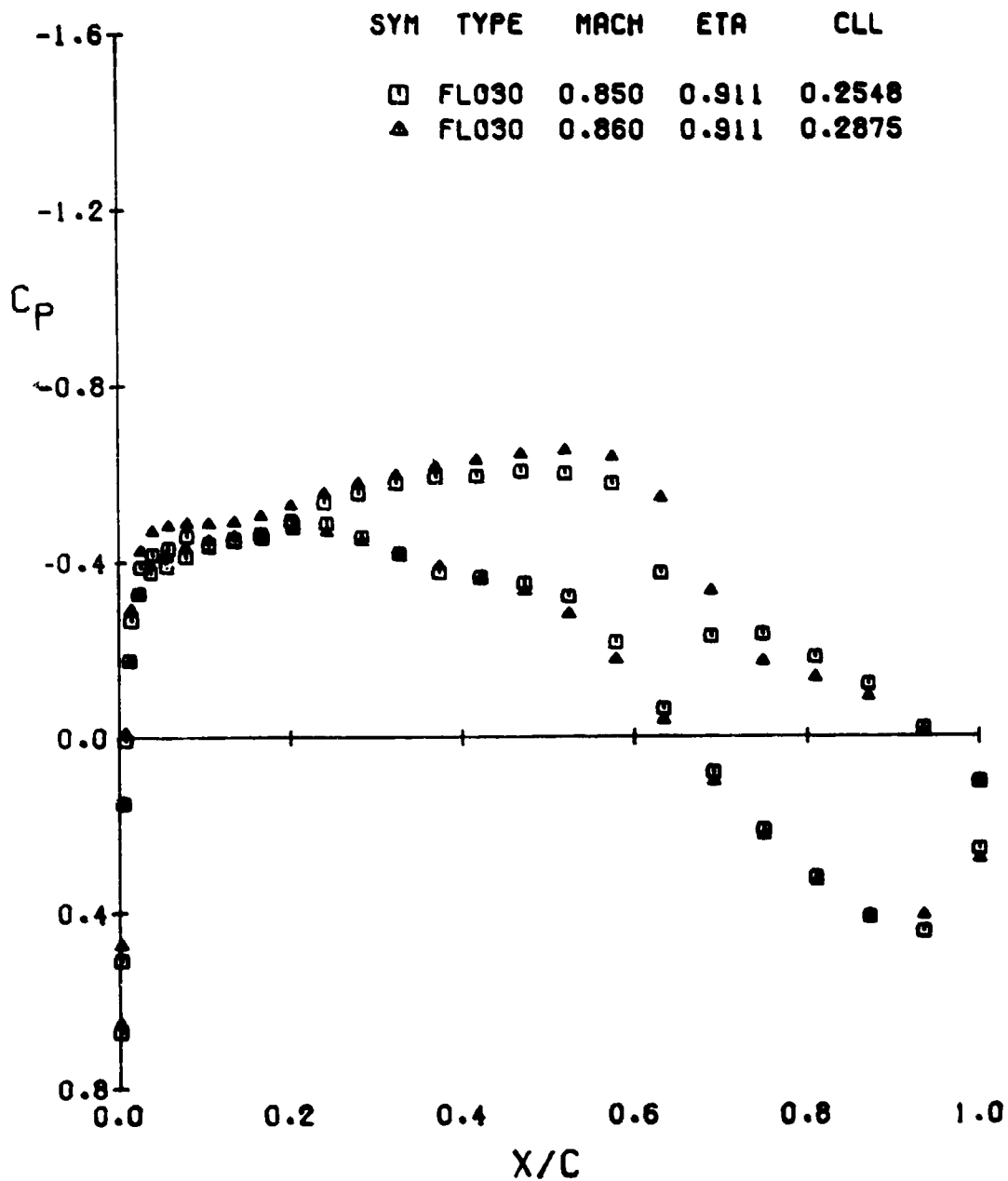
BODY/WING NO.2 T/C=.09



(C) ETA = 0.644

FIGURE 14.- CONTINUED

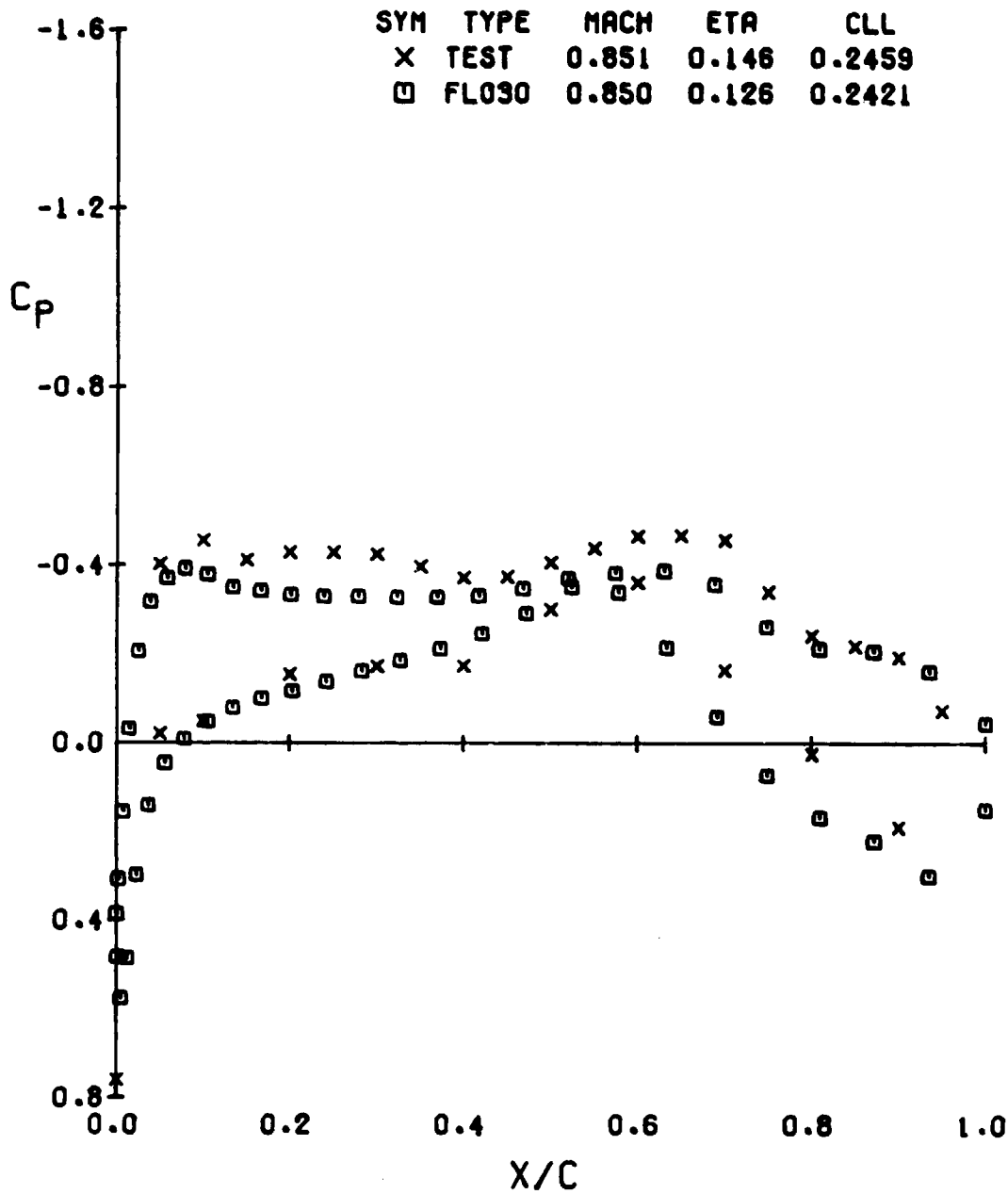
BODY/WING NO.2 T/C=.09



(O) ETA = 0.911

FIGURE 14.- CONCLUDED

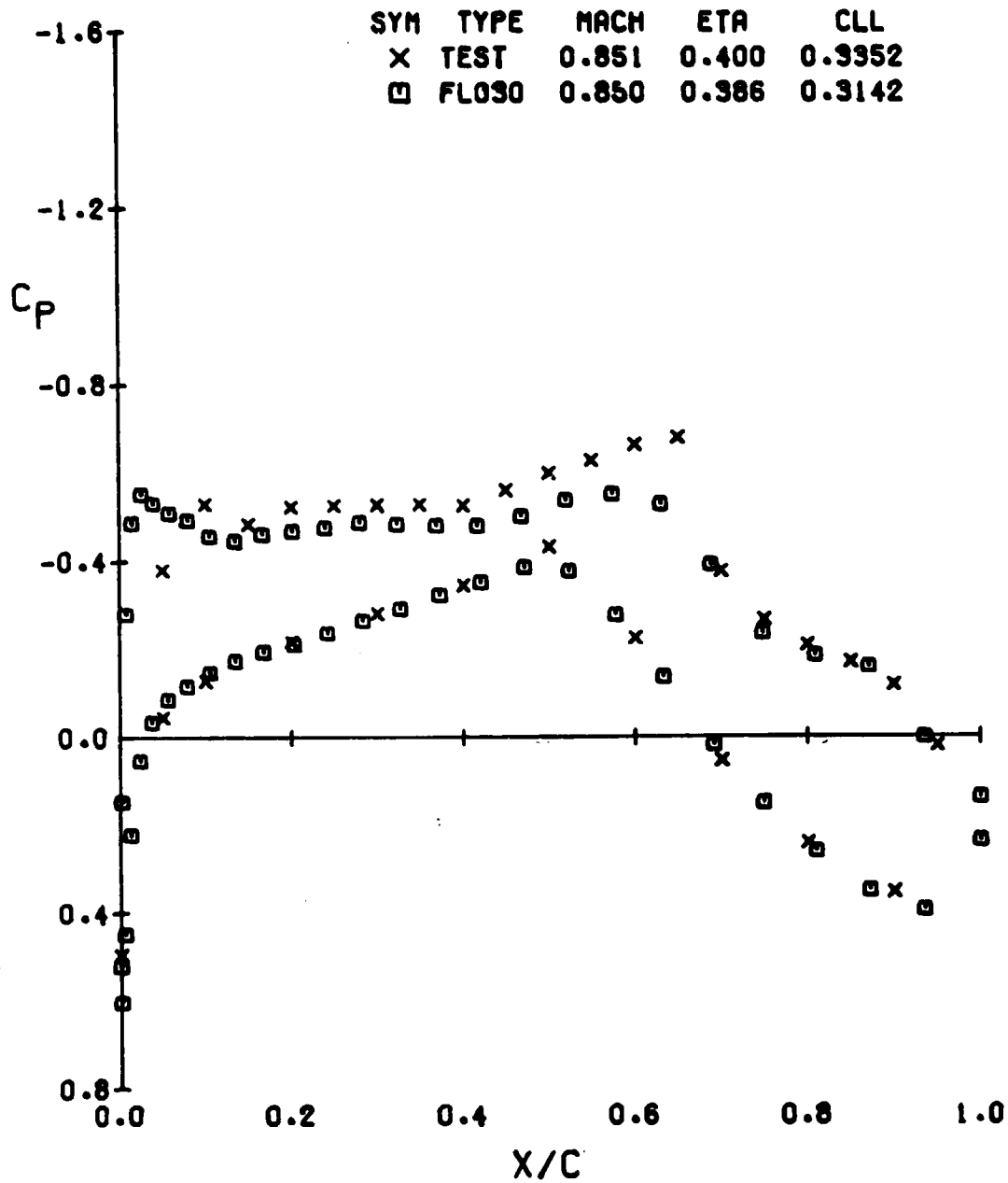
BODY/WING NO.1 T/C=.12



(A) ETA = 0.146

FIGURE 15.-- COMPARISON OF FLO30 WING/BODY PRESSURES WITH EXPERIMENT
 ALPHA = 4.68 CL(EXP) = .34

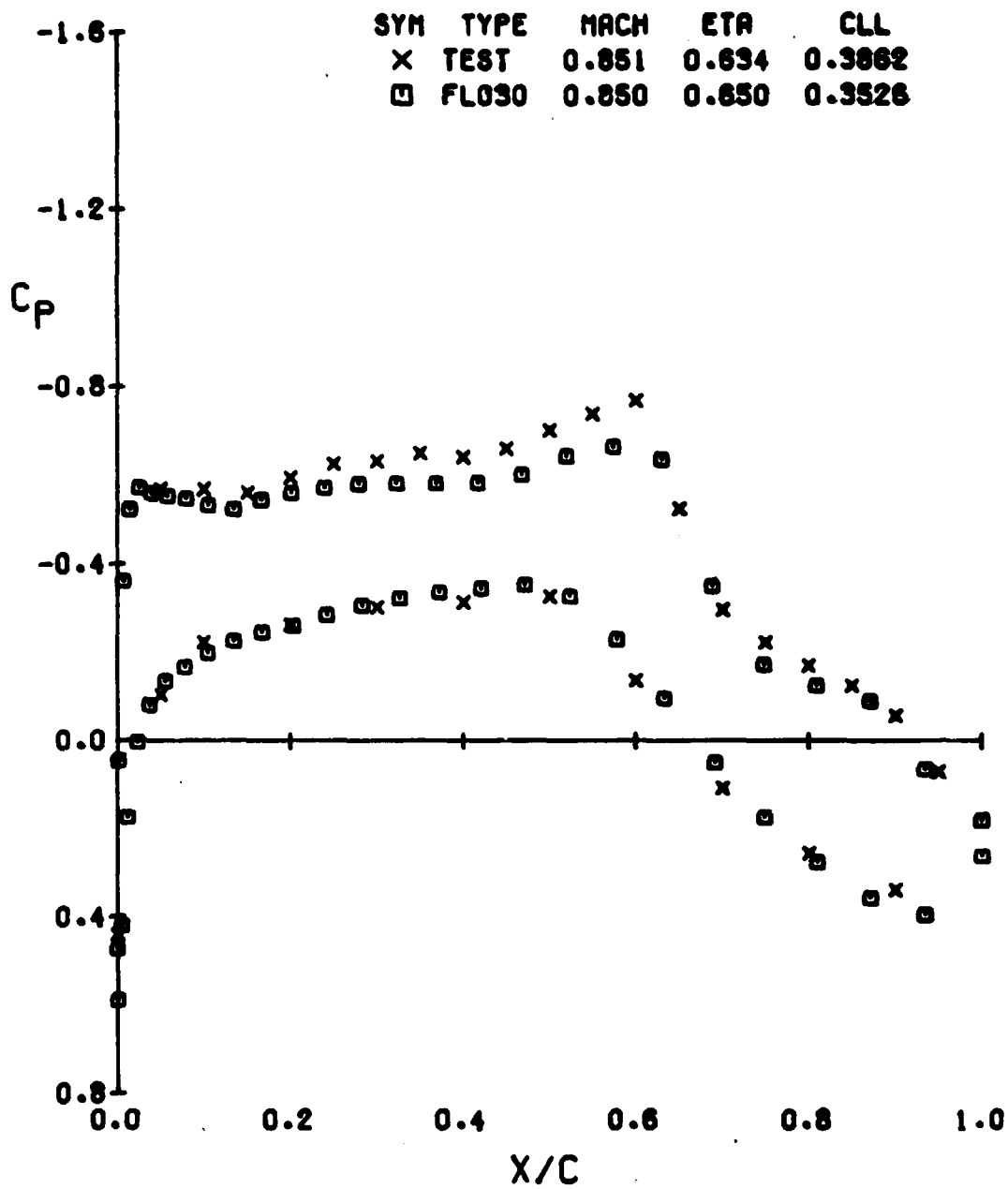
BODY/WING NO.1 T/C=.12



(B) ETA = 0.400

FIGURE 15.- CONTINUED

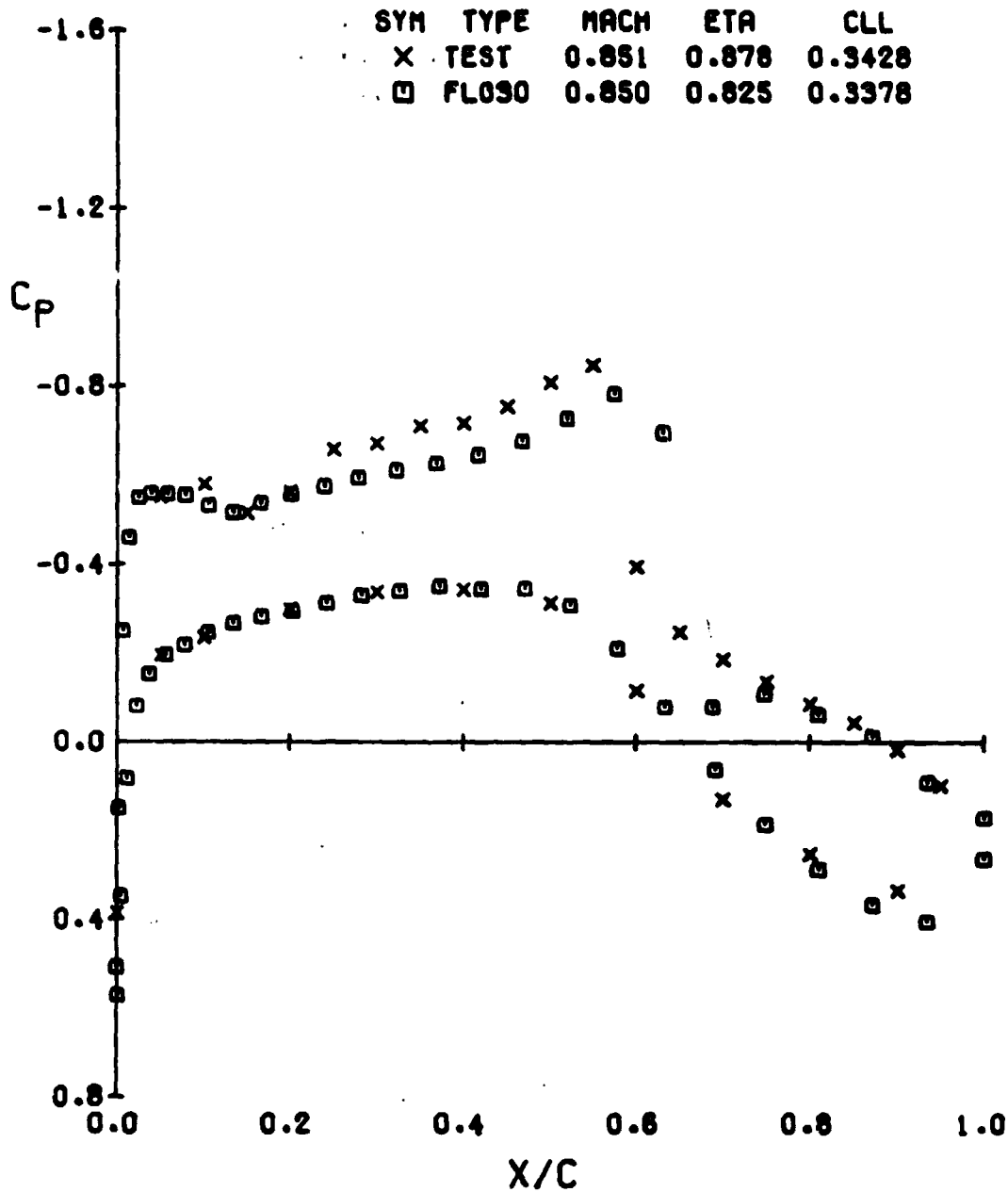
BODY/WING NO.1 T/C=.12



(C) ETA = 0.634

FIGURE 15.- CONTINUED

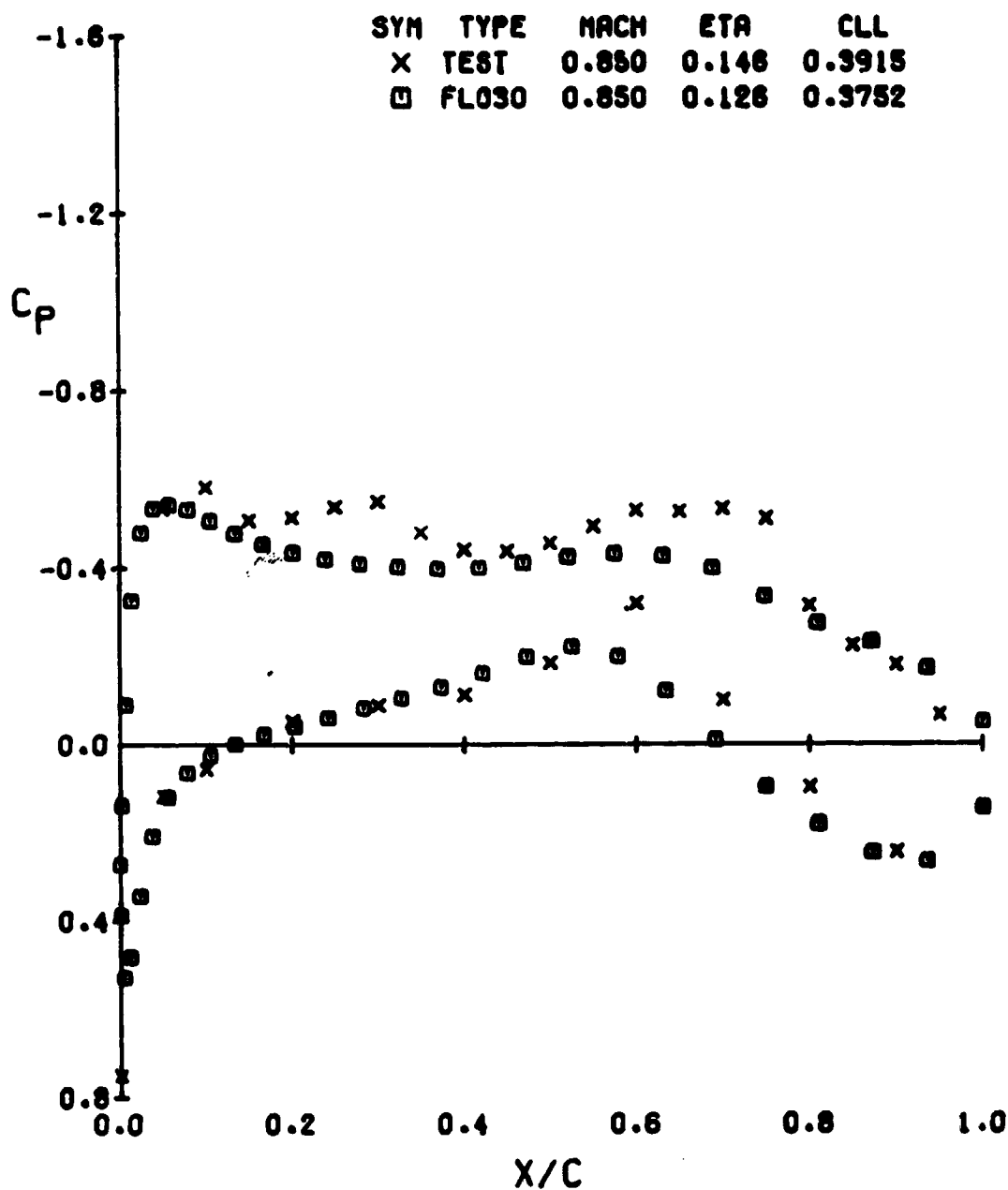
BODY/WING NO.1 T/C=.12



(O) ETA = 0.878

FIGURE 15.- CONCLUDED

BODY/WING NO.1 T/C=.12

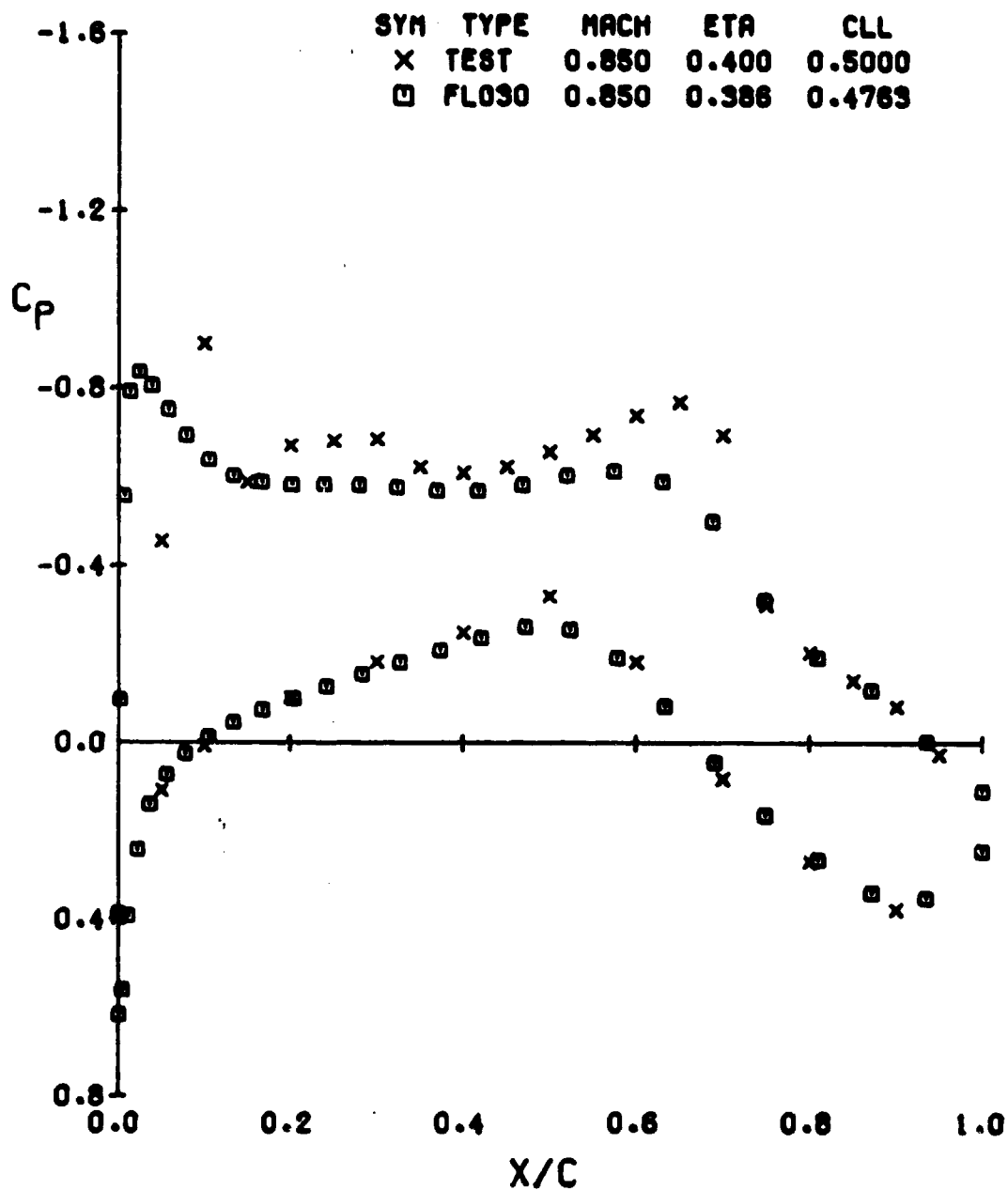


(A) ETA = 0.146

FIGURE 16.- COMPARISON OF FLO30 WING/BODY PRESSURES WITH EXPERIMENT

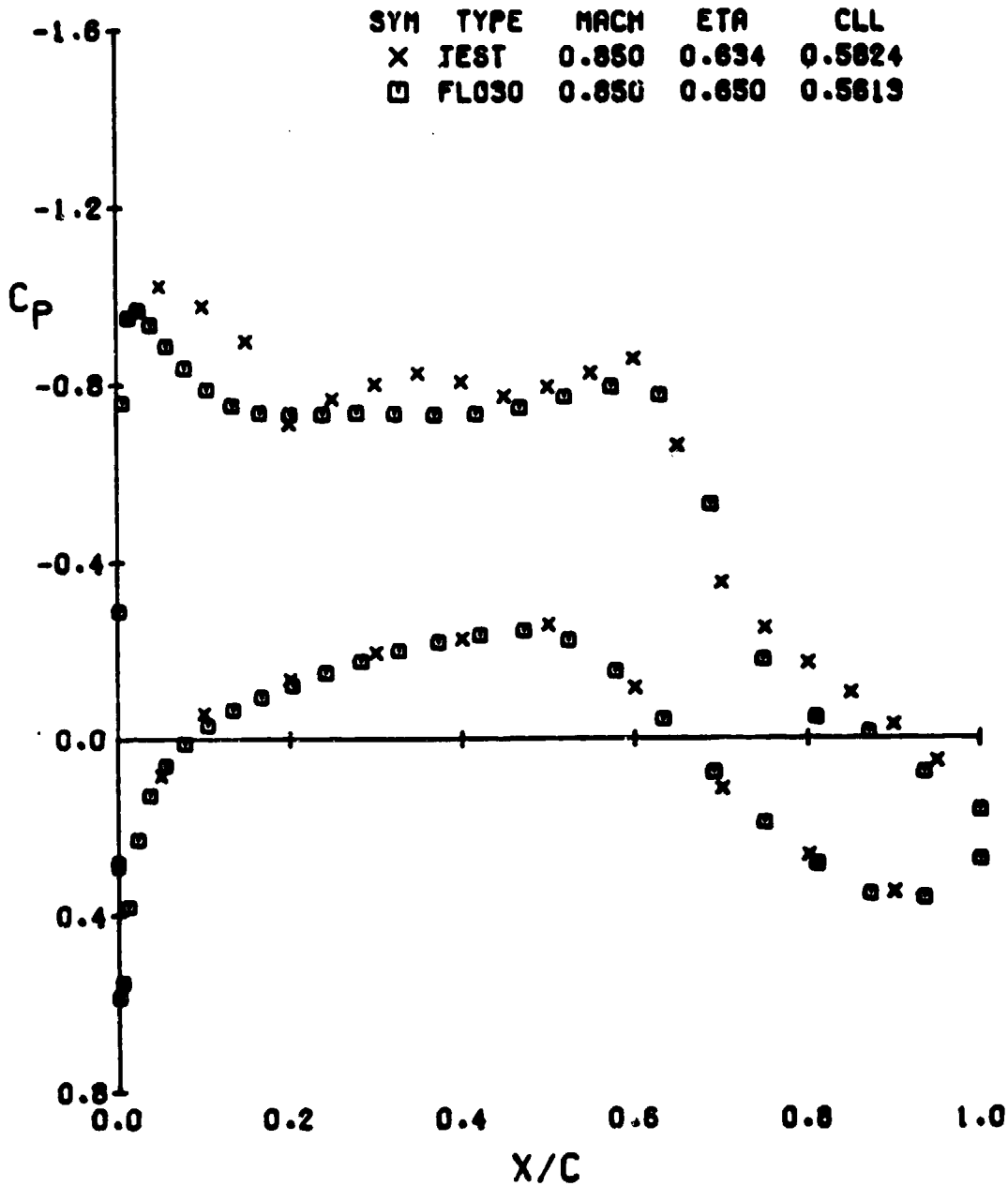
ALPHA = 6.52 CL(EXP) = .515

BODY/WING NO.1 T/C=.12



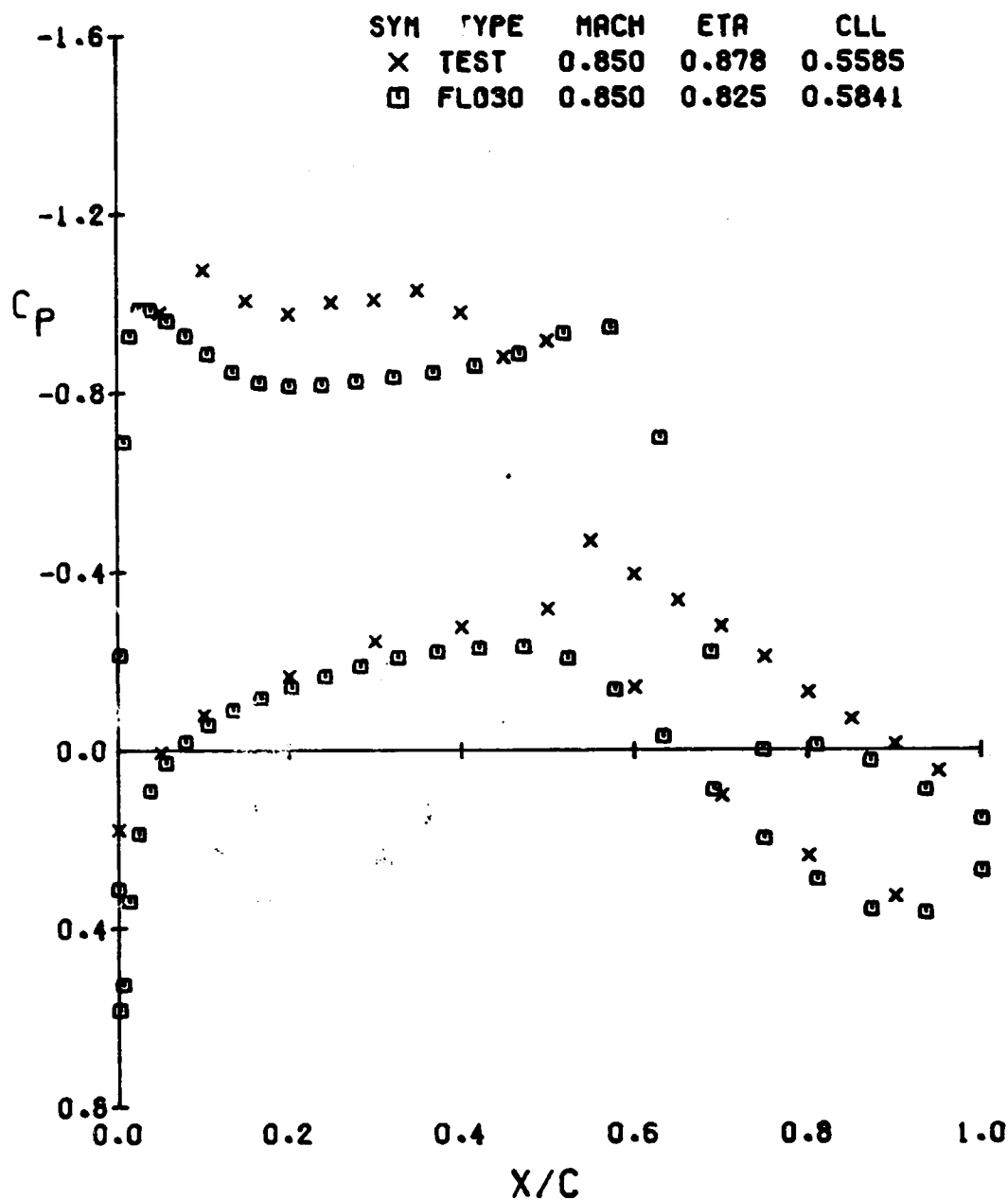
(B) ETA = 0.400
FIGURE 16.- CONTINUED

BODY/WING NO.1 T/C=.12



(C) ETA = 0.634
 FIGURE 16.- CONTINUED

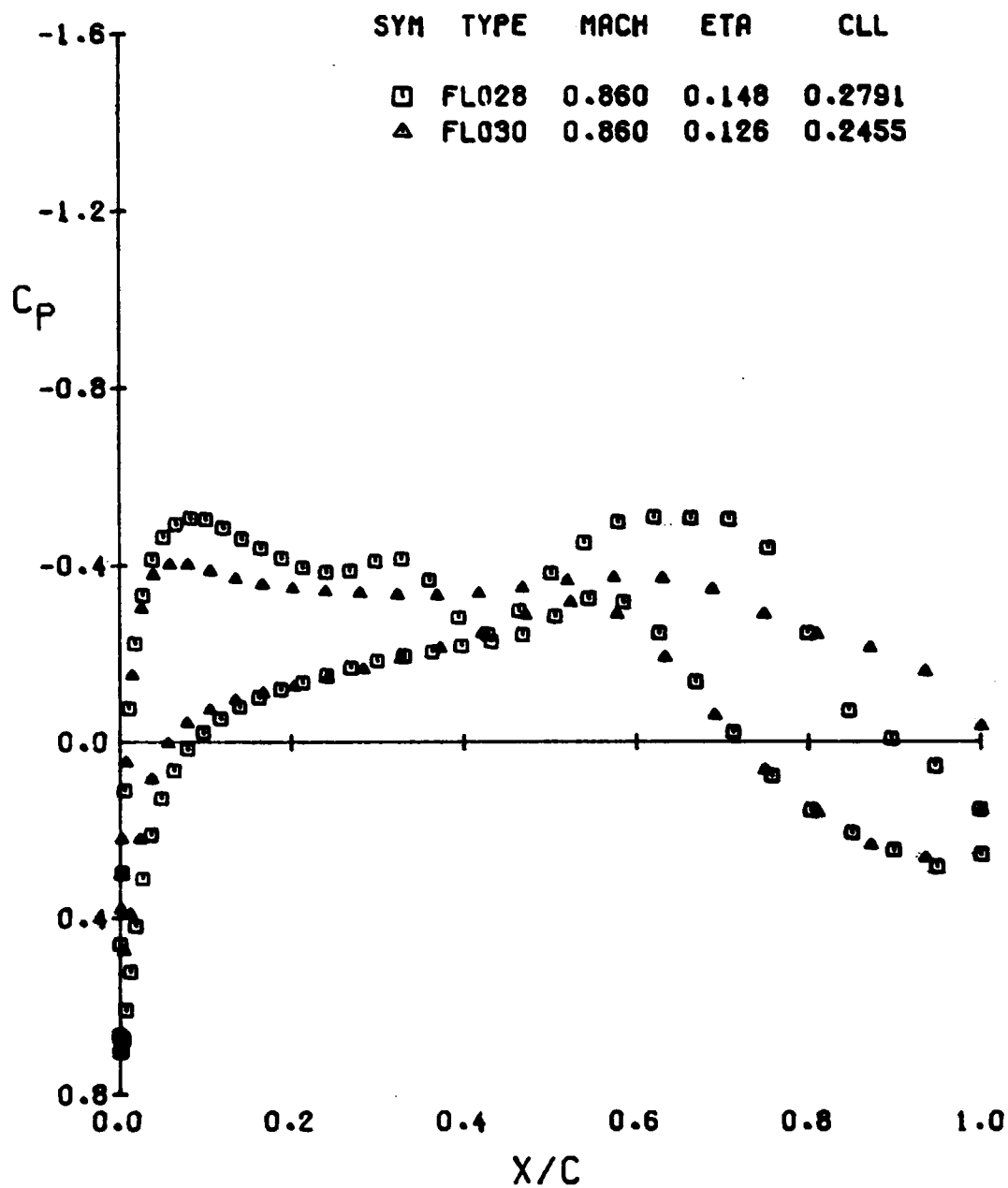
BODY/WING NO.1 T/C=.12



(O) ETA = 0.878

FIGURE 16.- CONCLUDED

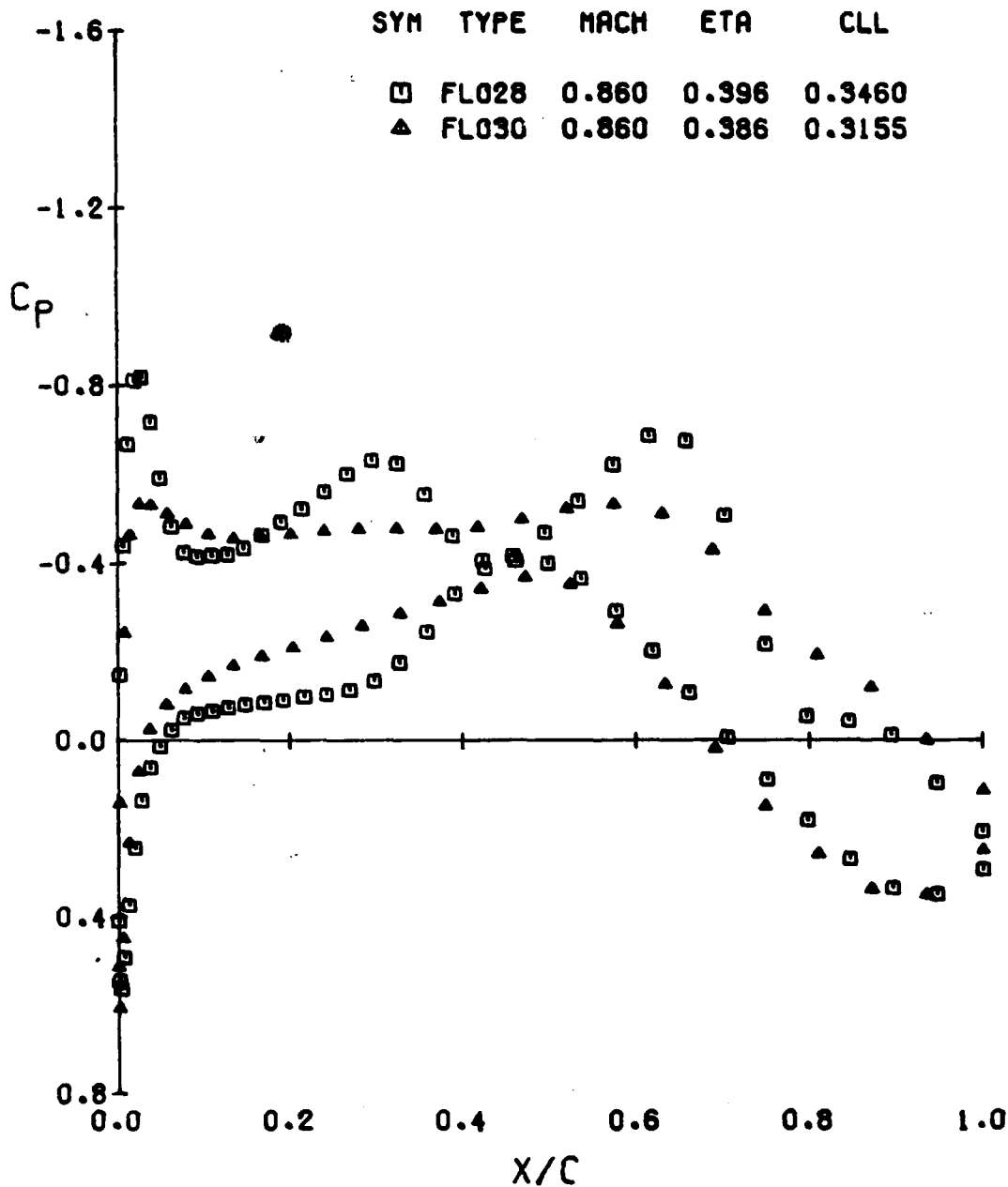
BODY/WING NO.1 T/C=.12



(A) ETA = 0.148

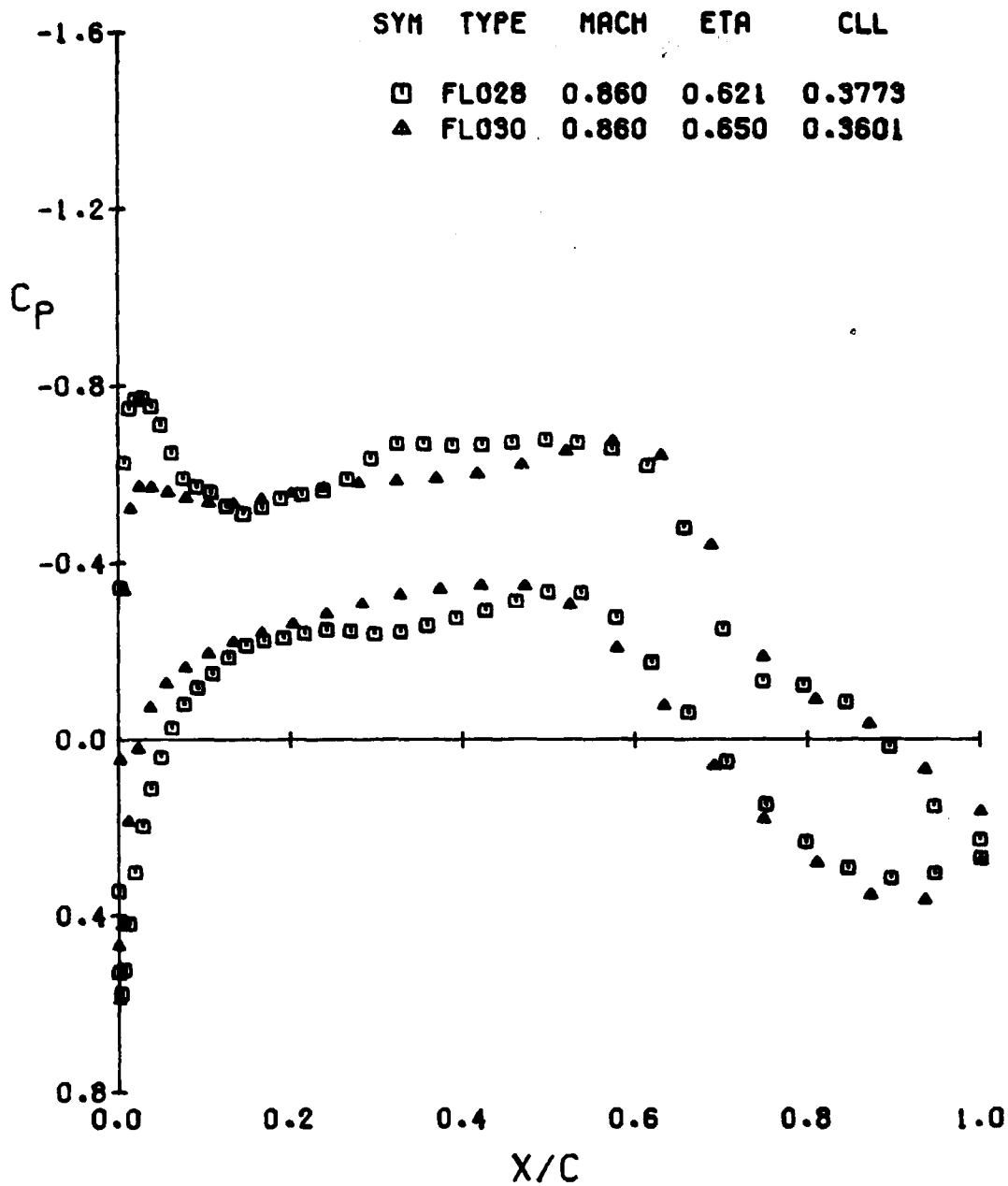
FIGURE 17.- COMPARISON OF FL028 AND FL030 SOLUTIONS

BODY/WING NO.1 T/C=.12



(B) ETA = 0.396
FIGURE 17.- CONTINUED

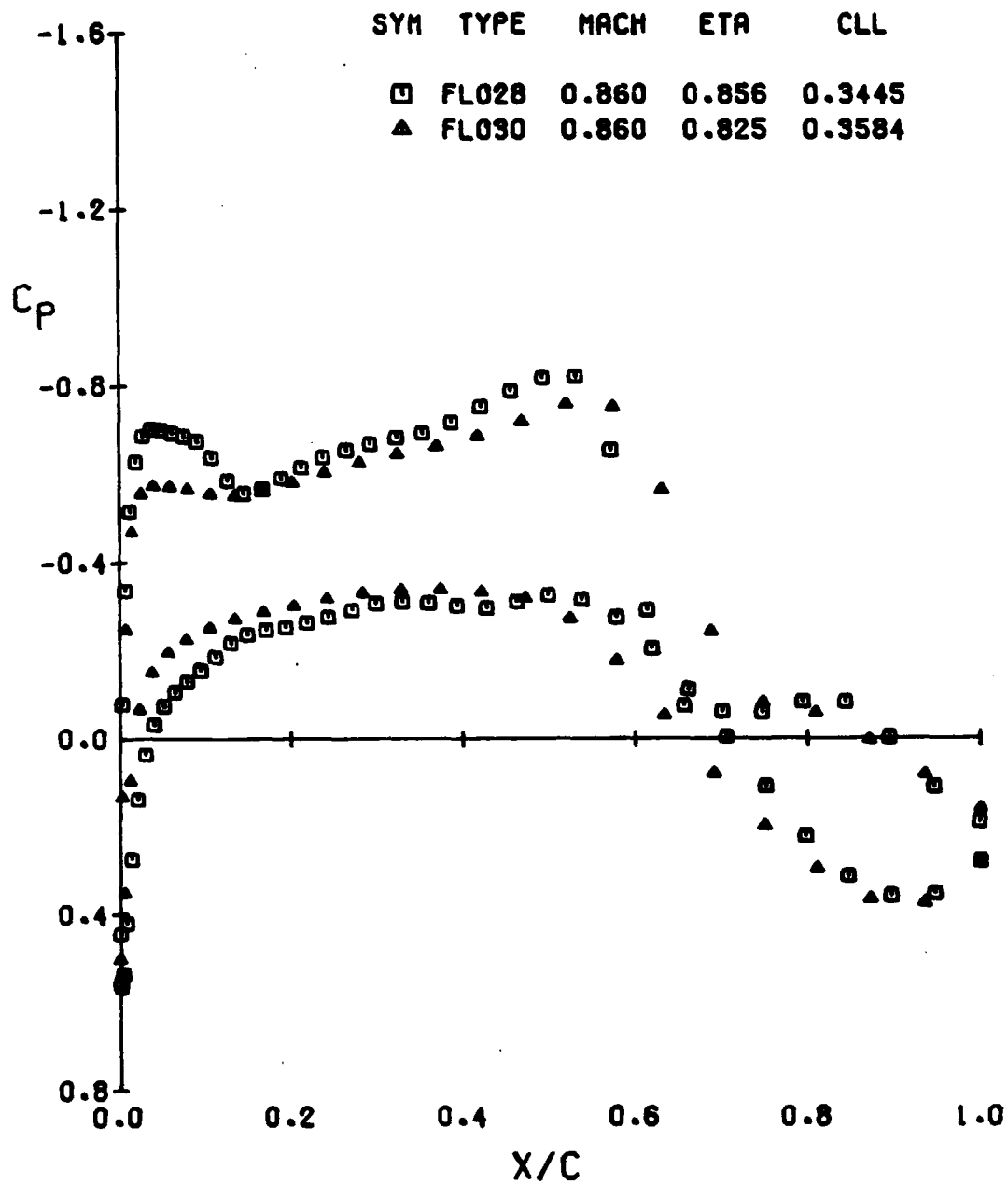
BODY/WING NO.1 T/C=.12



(C) ETA = 0.621

FIGURE 17.- CONTINUED

BODY/WING NO.1 T/C=.12

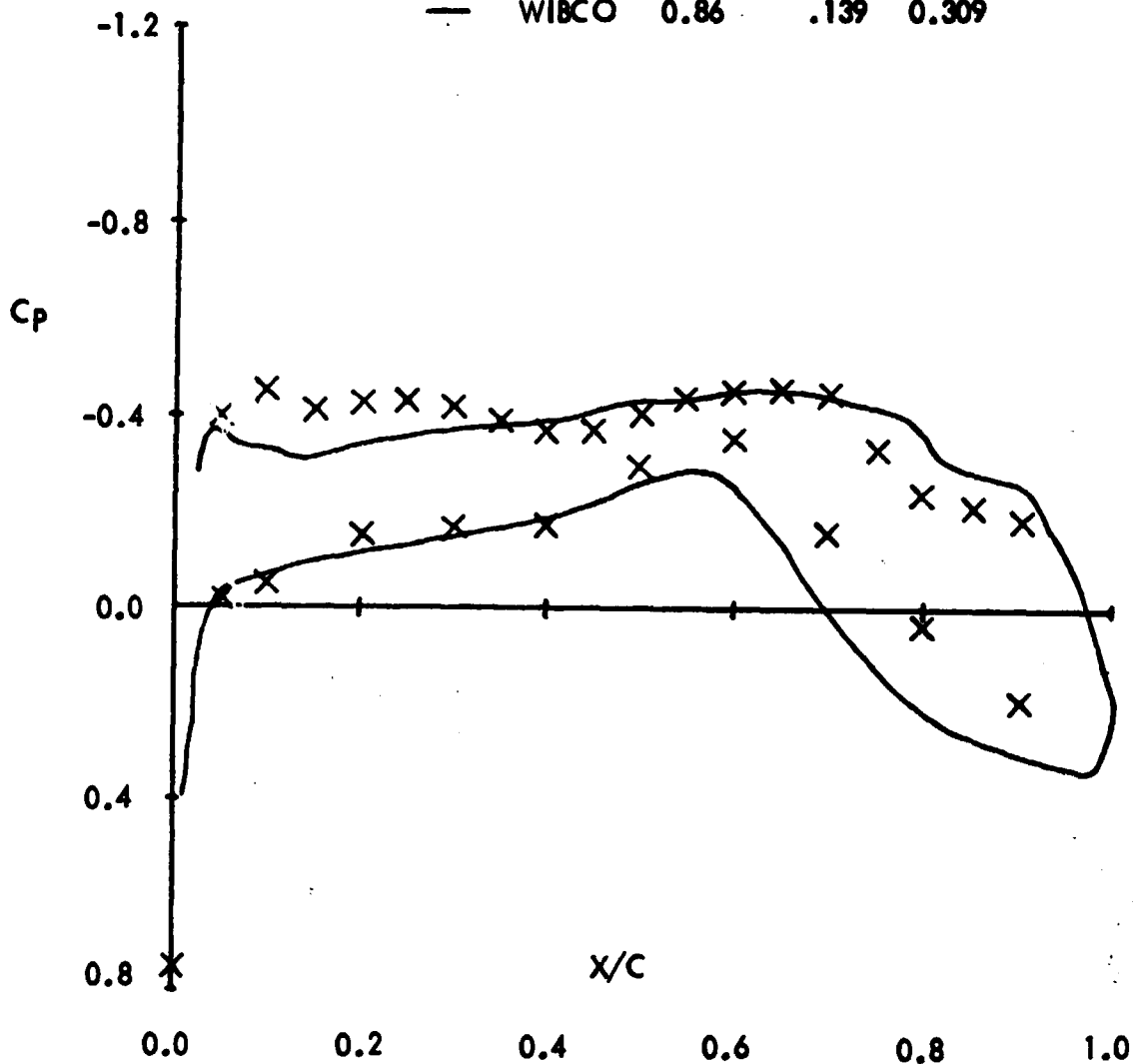


(O) ETA = 0.856

FIGURE 17.- CONCLUDED

WING NO. 1 $(t/c)_{MAX} = .12$

SYM	TYPE	MACH	ETA	CLL
X	TEST	0.851	.146	0.246
—	WIBCO	0.86	.139	0.309



(A) $\eta = 0.146$

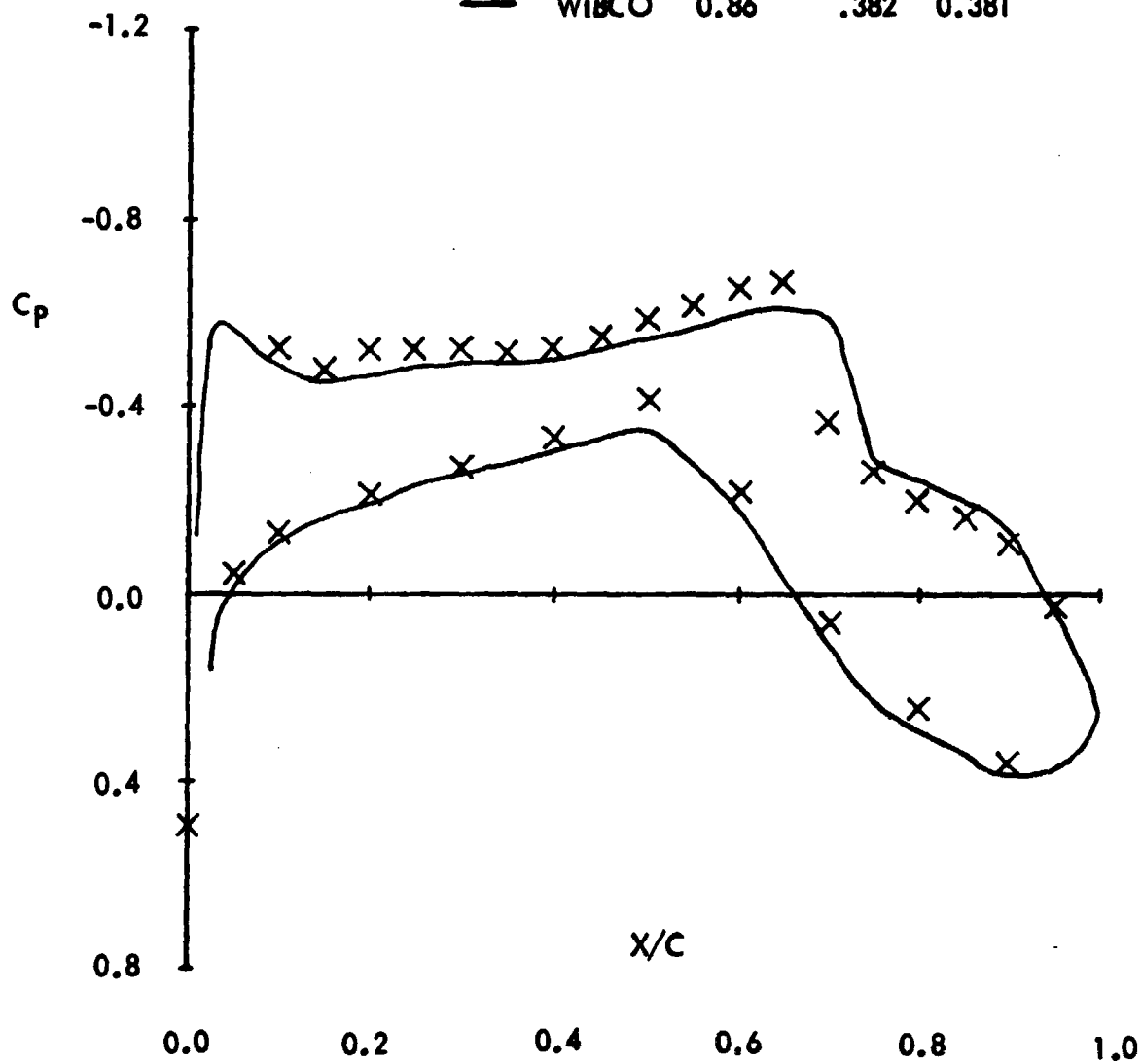
FIGURE 18. - COMPARISON OF WIBCO WING/BODY PRESSURES WITH EXPERIMENT

$\alpha = 4.68$

$C_L = .34$

WING NO. 1 $(t/c)_{MAX} = .12$

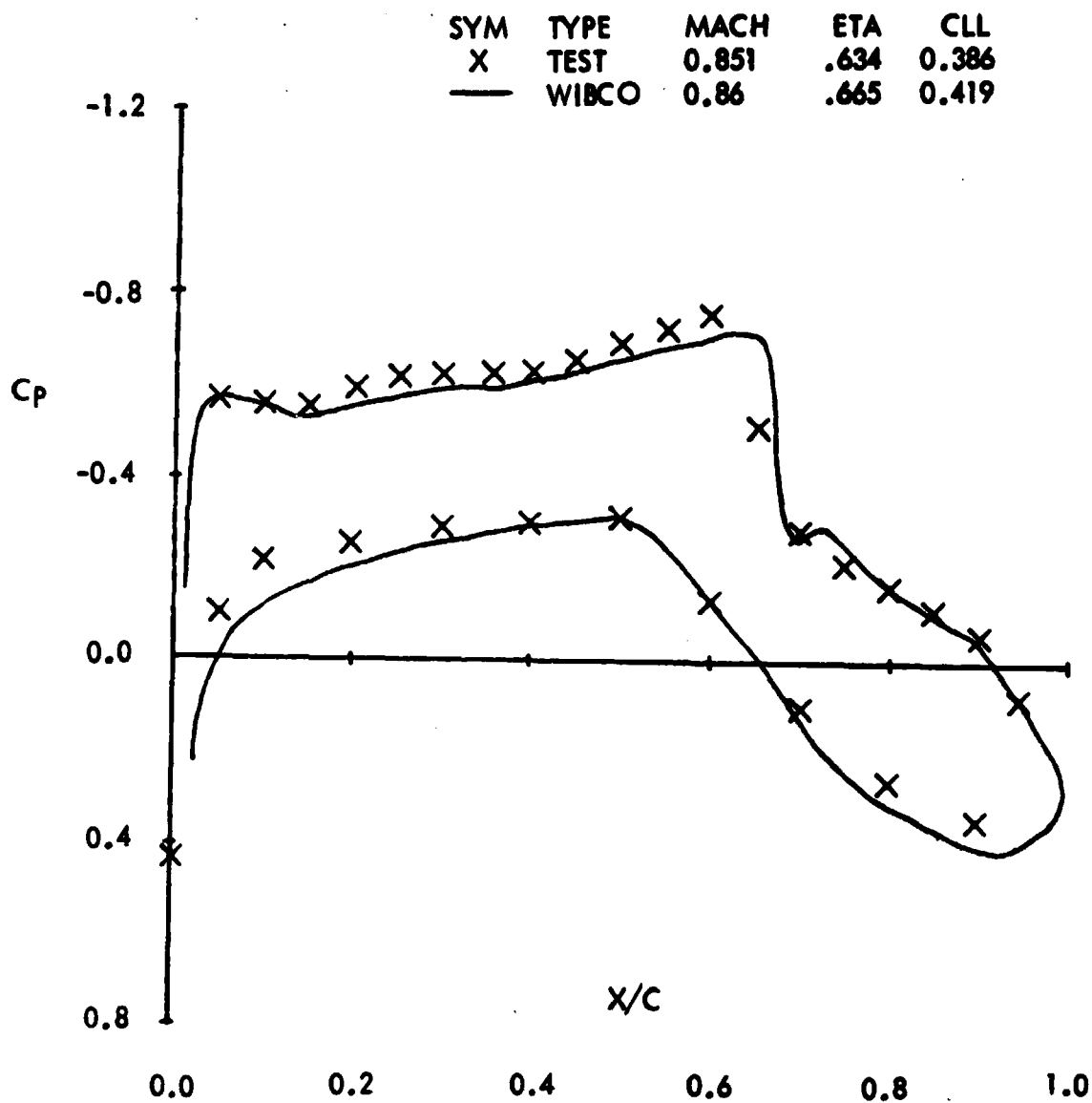
SYM	TYPE	MACH	ETA	CLL
X	TEST	0.851	.40	0.335
—	WIBCO	0.86	.382	0.381



(B) $ETA = 0.40$

FIGURE 18. - CONTINUED

WING NO. 1 $(t/c)_{MAX} = .12$

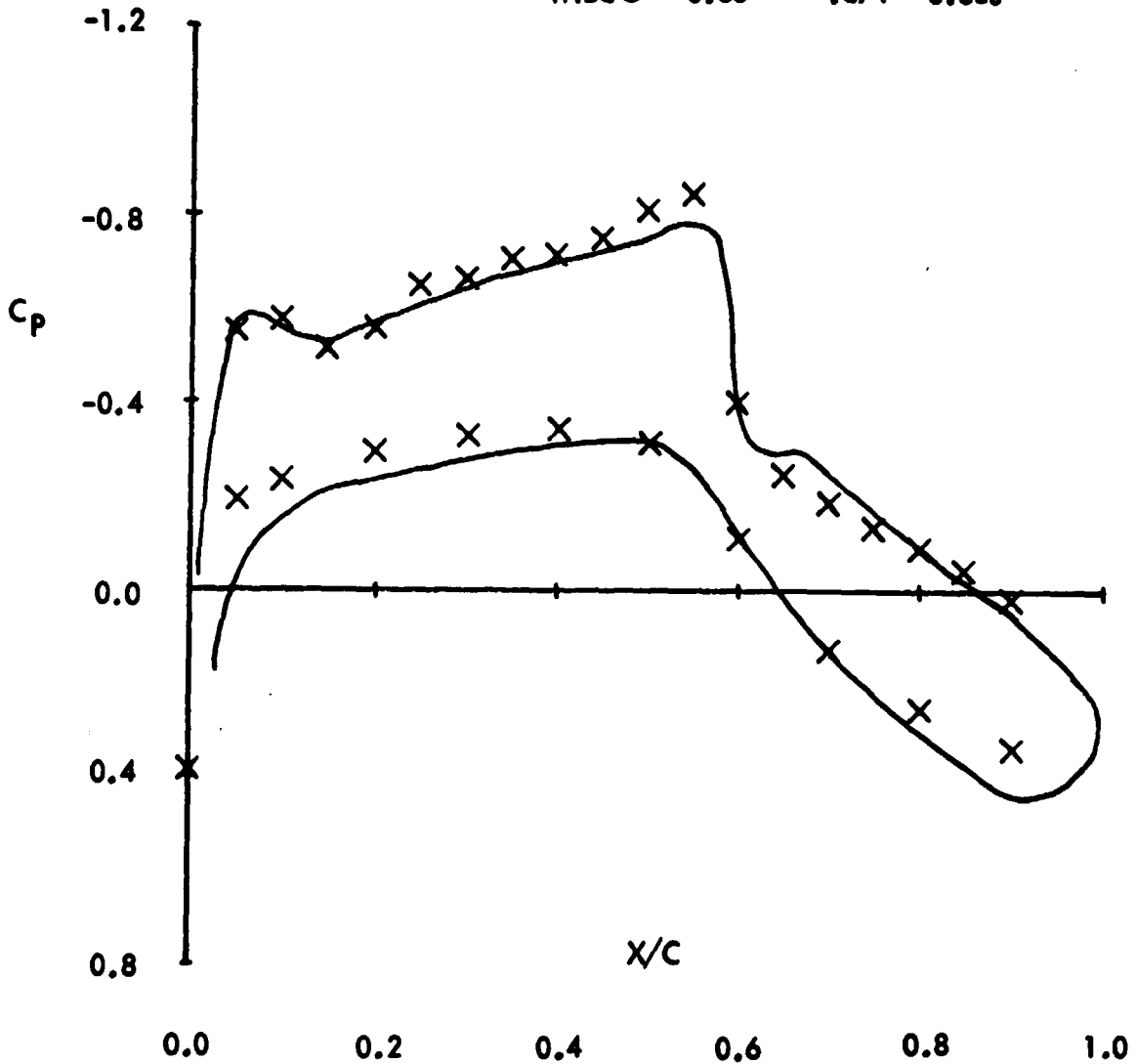


(C) $ETA = 0.634$

FIGURE 18. - CONTINUED

WING NO. 1 $(t/c)_{MAX} = .12$

SYM	TYPE	MACH	ETA	CLL
X	TEST	0.851	.878	0.343
—	WIBCO	0.86	.874	0.380

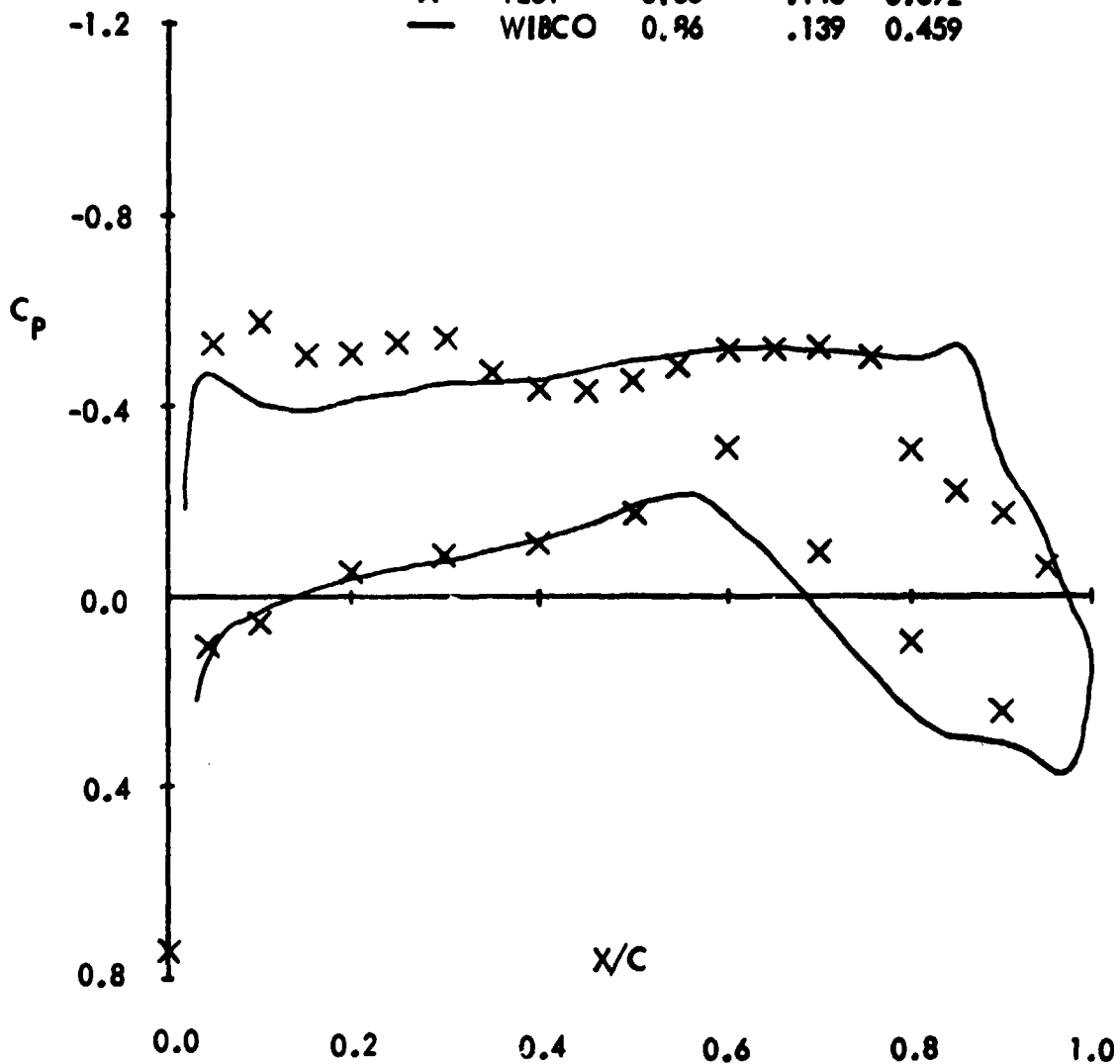


(D) $\eta = 0.878$

FIGURE 18. CONCLUDED

WING NO. 1 $(t/c)_{MAX} = .12$

SYM	TYPE	MACH	ETA	CLL
X	TEST	0.85	.146	0.392
—	WIBCO	0.96	.139	0.459



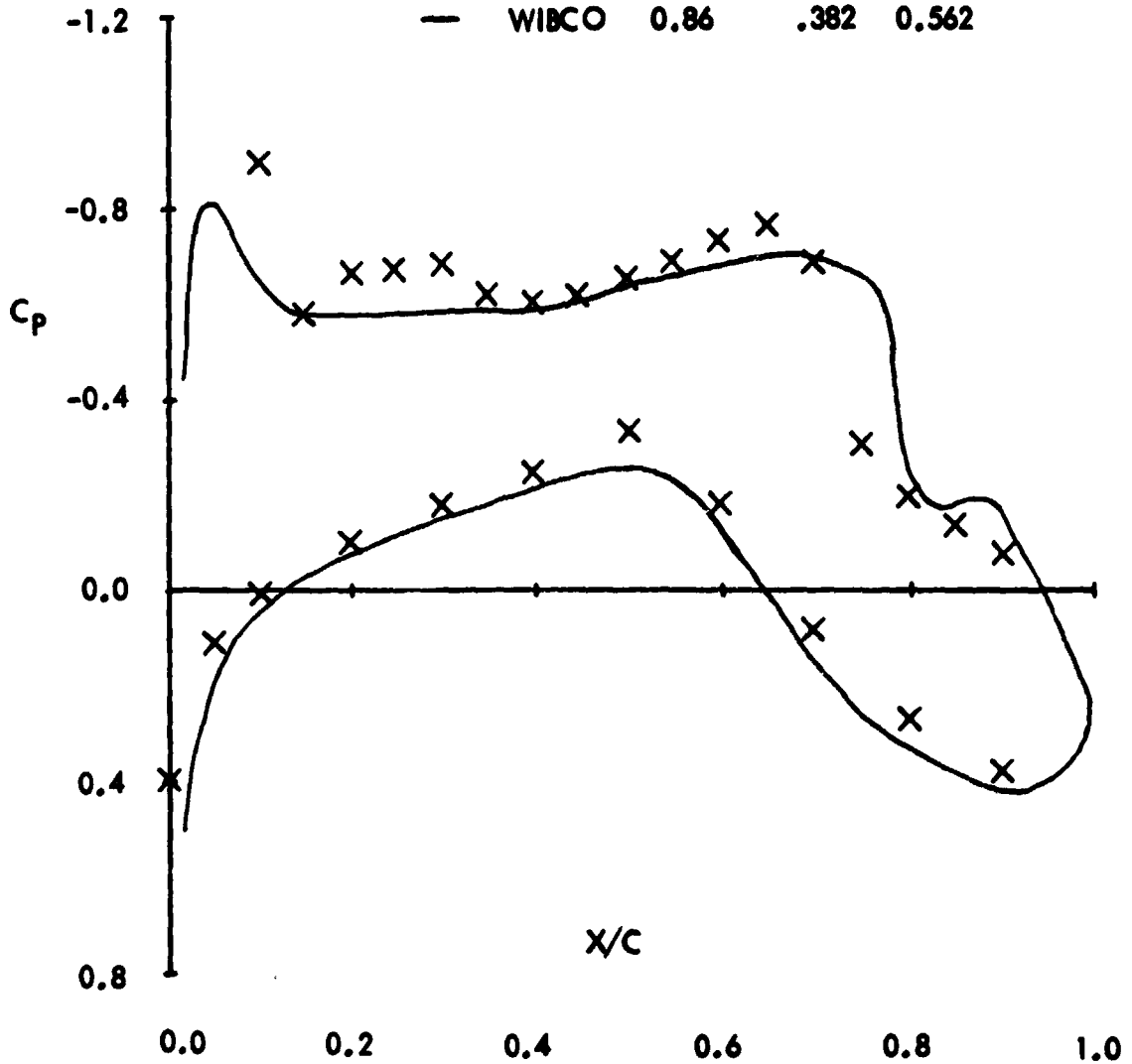
(A) $\eta = 0.146$

FIGURE 19. - COMPARISON OF WIBCO WING/BODY PRESSURES WITH EXPERIMENT

$\alpha = 6.52$ $C_L = .52$

WING NO. 1 $(t/c)_{MAX} = .12$

SYM	TYPE	MACH	ETA	CLL
X	TEST	0.85	.40	0.50
—	WIBCO	0.86	.382	0.562

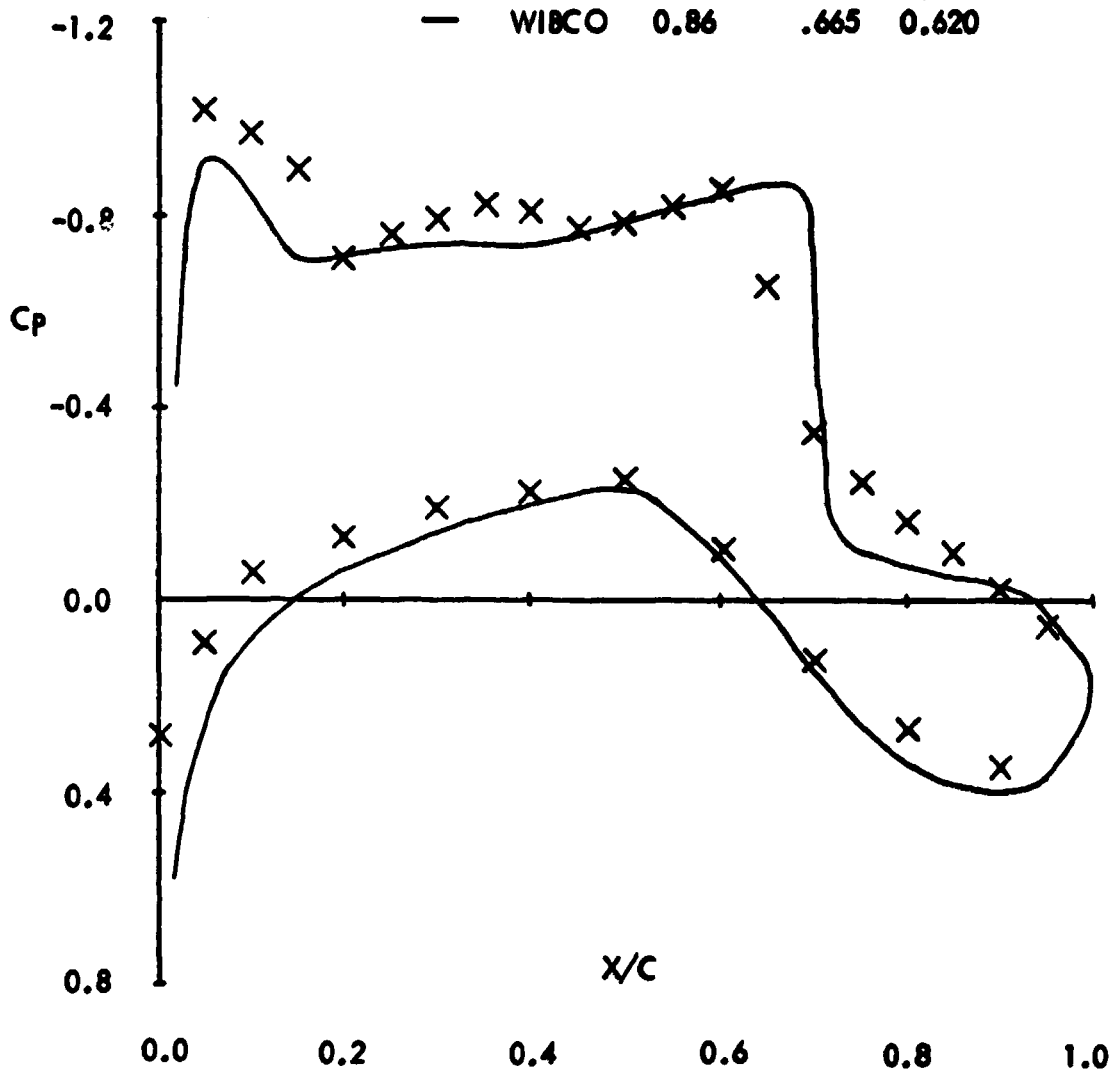


(B) $\eta = 0.40$

FIGURE 19. - CONTINUED

WING NO. 1 $(l/c)_{MAX} = .12$

SYM	TYPE	MACH	ETA	CLL
X	TEST	0.85	.634	0.582
—	WIBCO	0.86	.665	0.620

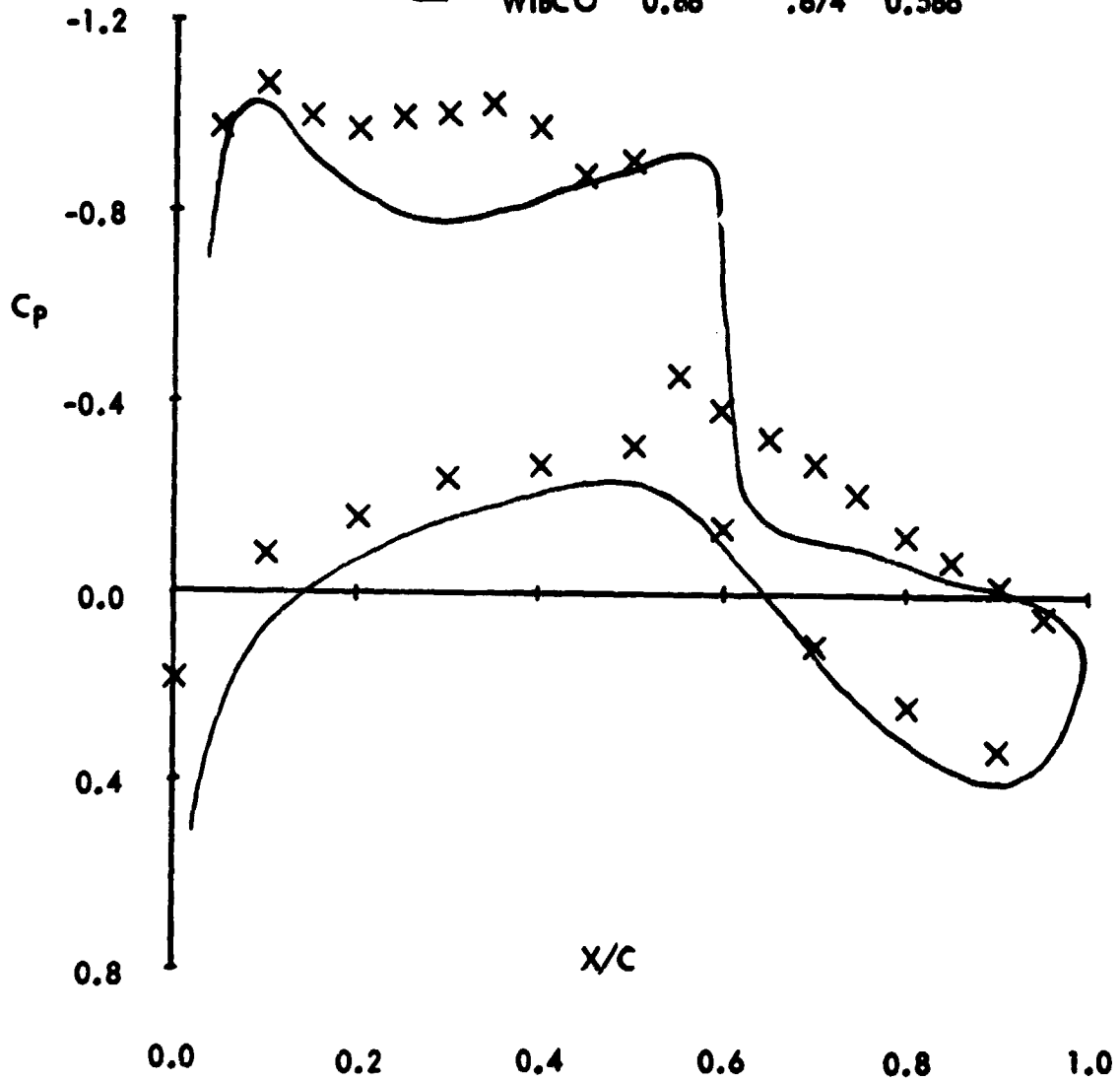


(C) $ETA = 0.634$

FIGURE 19. - CONTINUED

WING NO. 1 $(t/c)_{MAX} = .12$

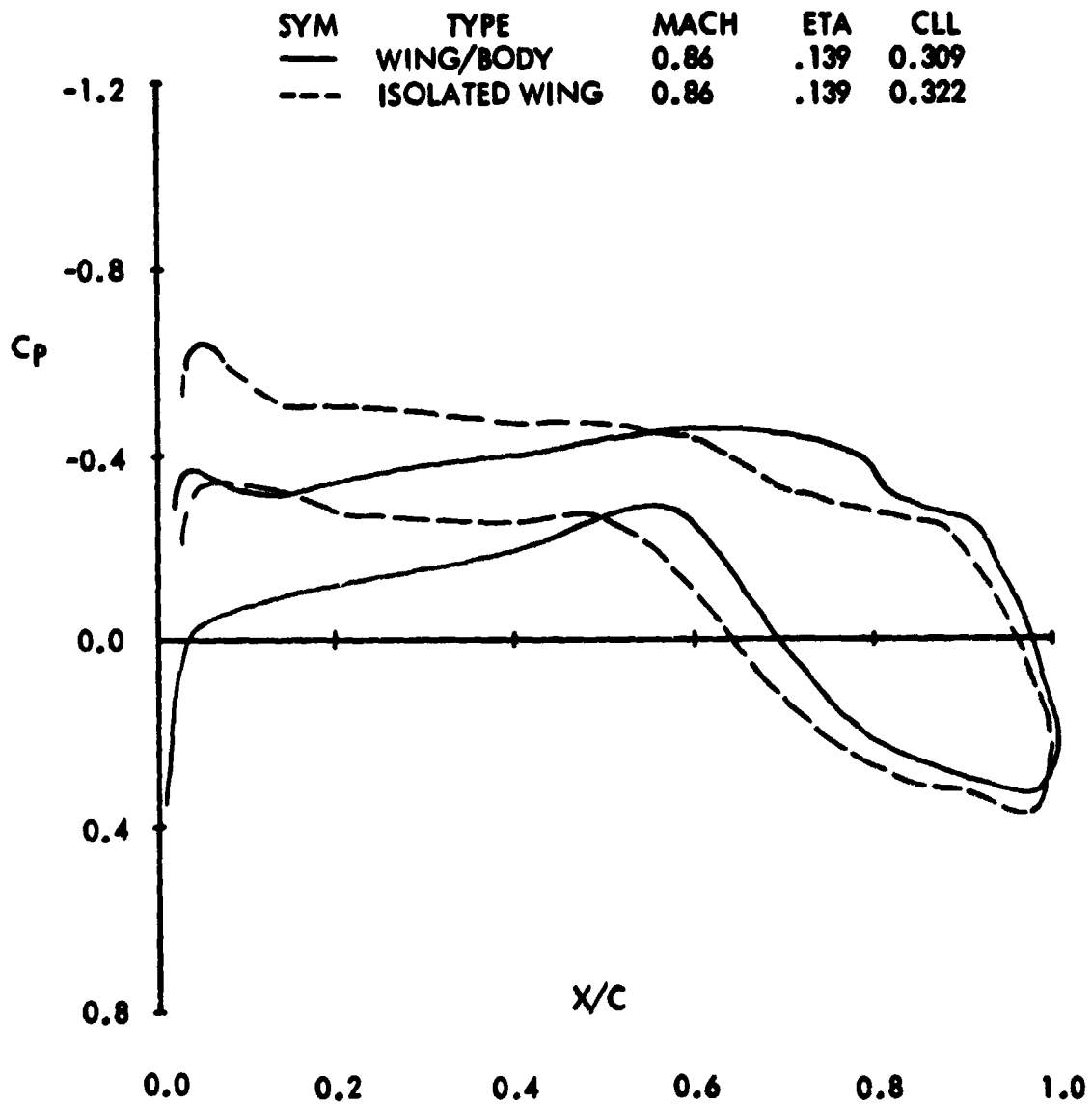
SYM	TYPE	MACH	ETA	CLI
X	TEST	0.85	.878	0.559
—	WIBCO	0.86	.874	0.586



(D) $ETA = 0.878$

FIGURE 19. - CONCLUDED

WING NO. 1 $(t/c)_{MAX} = .12$

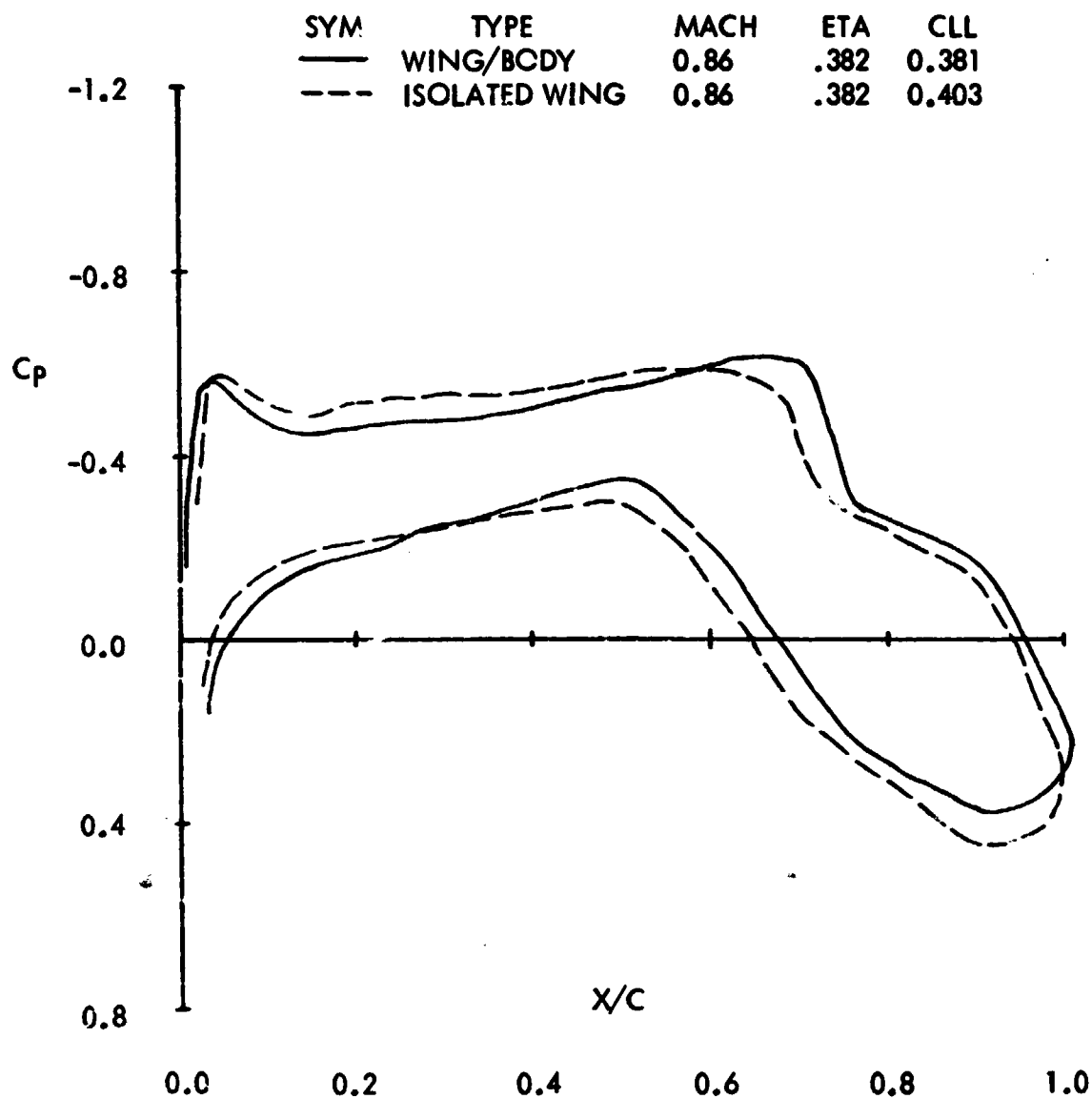


(A) $\text{ETA} = 0.146$

FIGURE 20. - COMPARISON OF WIBCO WING/BODY AND ISOLATED WING PRESSURES

$\text{ALPHA} = 4.68$

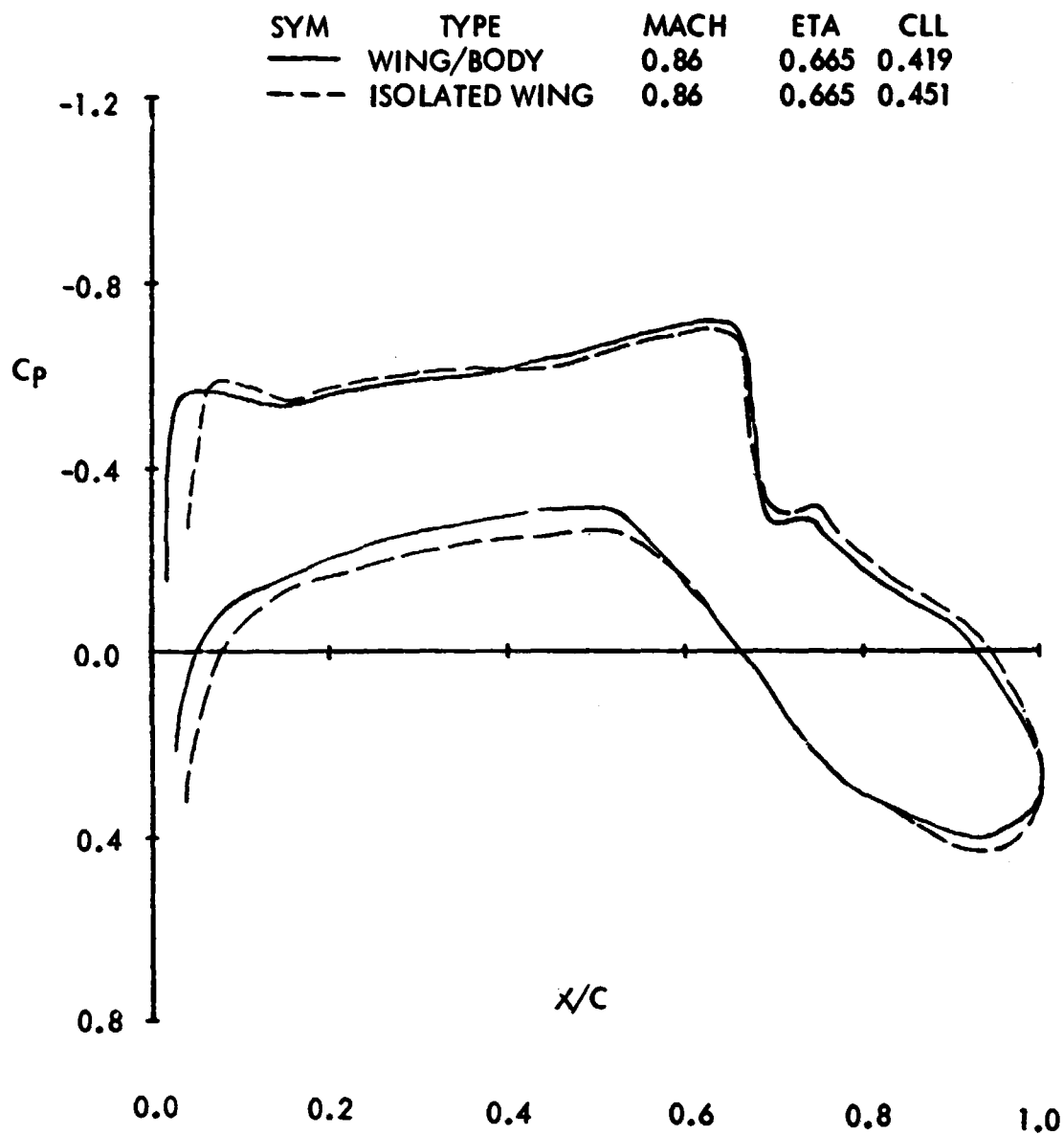
WING NO. 1 $(t/c)_{MAX} = .12$



(B) $\eta = 0.40$

FIGURE 20. - CONTINUED

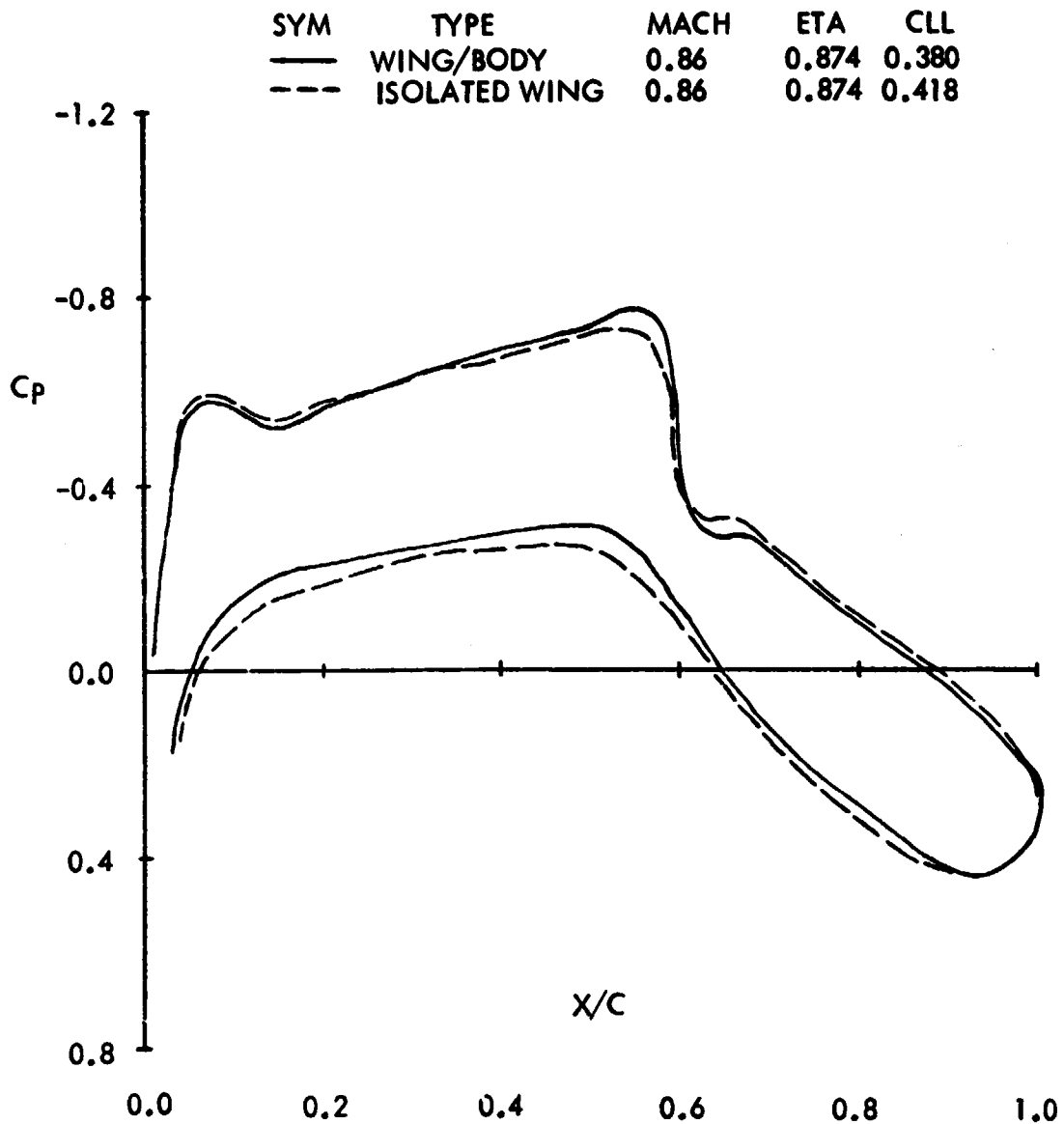
WING NO. 1 $(t/c)_{MAX} = .12$



(C) $ETA = 0.634$

FIGURE 20. - CONTINUED

WING NO. 1 $(t/c)_{MAX} = .12$



(D) $\eta = 0.878$

FIGURE 20. - CONCLUDED